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14. Abstract

In 1982, the Propulsion and Energetics Panel Working Group 17 endeavoured to investigate test techniques applied to solid rocket motors with metallized propellants. It was stated that there is a large number of related test facilities in existence with a large variety of technologies applied and test possibilities offered.

It was decided to compile a register showing the major altitude test facilities with their characteristic data. Emphasis was put on facilities capable of performing research and development tests.

This AGARDograph was prepared at the request of the Propulsion and Energetics Panel of AGARD.

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ADVISORY GROUP FOR AEROSPACE RESEARCH & DEVELOPMENT

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Rocket Altitude Test Facilities Register

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ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT

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compiled by

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France

MARCH 1987

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PREFACE

In the field of rocket propulsion (both liquid and solid), the simulation of the environmental parameters in static tests is a very important goal, but one which is frequently difficult to achieve; if the temperature conditions can be obtained easily, the same is not true of the acceleration (stationary or unstationary) loads. Another important parameter is the external pressure, especially if the motor is designed for operating at high altitude; this is obviously the case for the upper stages of ballistic missiles or satellite launchers and for space-boost motors.

Some examples of vacuum-related technical problems are given below:

— the question of ignition (or re-ignition) under vacuum (for liquid, especially cryogenic, propellants),

— thermal problems related to the lack of natural convection,

— mechanical behaviour of exit-cones (especially hot cones using carbon-carbon materials),

— operation of extendible exit-cones during extension on the plume,

— precise measurement of specific impulse under vacuum conditions.

Working Group No.17 of the AGARD Propulsion and Energetics Panel was investigating this last point when it appeared that it would be very interesting to create an AGARDograph gathering all the data related to altitude simulation facilities currently existing in the western world.

This is precisely the object of the excellent work undertaken by Mr Ducasse. One can be sure that this document will be very useful to all the engineers involved in questions dealing with altitude simulation; for the test engineer in charge of the design of a new facility as well as for the programme manager looking for a test centre capable of testing his motor under altitude simulation before the first flight test, which is a way of greatly reducing the hazards of development.

* * *

Dans le domaine de la propulsion par moteur-fusée (à ergols liquides ou solides) la simulation dans les tirs au banc de l'environnement réel du moteur est un objectif important, mais bien souvent difficile à obtenir; si les conditions de températures peuvent être facilement obtenues, il n'est est pas de même des charges d'accélération (stationnaires et instationnaires). Un autre paramètre important est la pression externe, surtout si le moteur est conçu pour fonctionner en haute altitude; c'est évidemment le cas des propulseurs pour étages supérieurs de missiles balistiques ou de lanceurs, de moteurs de transferts d'orbites pour les satellites.

On peut donner l'exemple de quelques problèmes techniques éventuels liés à l'existence du vide:

— problèmes liés à l'allumage ou au réallumage sous vide (pour les ergols liquides surtout, et notamment cryogéniques).

— problèmes thermiques liés à l'absence de refroidissement extérieur par convection naturelle.

— tenue mécanique des divergents (surtout ceux qui fonctionnent à température élevée et qui utilisent des matériaux carbone-carbone).

— mesure précise de l'impulsion spécifique sous vide.

C'est justement en examinant ce dernier point que le groupe de travail no.17 du PEP AGARD a eu l'idée de proposer la constitution d'un AGARDographie rassemblant les données relatives aux installations de simulation d'altitude existant dans le monde occidental.

C'est précisément l'objet de l'excellent travail réalisé par Mr Ducasse.

On peut être sûr que ce document sera très utile à tous les ingénieurs en prise avec un problème de simulation d'altitude, qu'au responsable de programme cherchant un centre d'essai capable de tester le fonctionnement de son moteur en simulation d'altitude avant le premier tir en vol, ce qui est de nature à réduire considérablement les aléas d'un programme.

l'Ingénieur en Chef de l'Armement
D.REYDELLET

CONTENTS

1. AN OVERVIEW OF SIMULATED ALTITUDE FACILITY CONCEPTS

1.1 GENERAL CONCEPTS

1.1.1 The aims

Flight conditions:

By altitude simulation, the reader should understand low, or very low, test article environmental pressure.

Generally, a simulated altitude of 33 km/100,000 ft is taken as a standard. But above all, in the future a higher altitude may be required. Such environmental conditions are necessary to perform tests concerning:

— aerodynamic behaviour of nozzles,
— ignition and re-ignition, especially for liquid rocket motors,
— heat transfer, especially after firing.

In a few cases, other flight conditions are simulated, mostly for space applications such as low temperature or solar radiation.

High area ratio nozzle testing

At high altitude, ambient pressure is comparatively low (fig.1), so it is possible to expand the combustion gases until nozzle exit static pressure is in relation with ambient pressure:

$P_{NE} > 0.6\ P_A$

where: P_{NE} is the nozzle exit static pressure
P_A is the ambient pressure
(when $P_{NE} = P_A$, the nozzle is adapted).

The ratio P_{NE}/P_C is in relation to nozzle expansion ratio ($\Sigma = A_{NE}/A^*$) as shown in fig.2:

P_c : combustion pressure,
A_{NE} : nozzle exit area,
A^* : nozzle throat area.

The higher the Σ, the better is the performance, similar to thrust coefficient C_F as shown in fig.3.

The nozzle exit static pressure can also be very low, even if expansion ratio is not very high, when the chamber pressure is comparatively low (1.0 MPa $\neq$ 150 psia), which is the case with liquid rocket motors.

At sea level, pressure is too high to permit full flow in nozzle, as shown in fig.4: instabilities in gas flow, or non-nominal vibrations appear, so it is impossible to evaluate performance, and the nozzle can be damaged by flow separation.

Extendible exit cone nozzles

If a nozzle has a high expansion ratio, it is also very long, its half-angle remaining about 20°. In missile and space systems, motors must be sized to meet very strict envelope constraints. For that reason, extendible exit cones (EEC) nozzles have been developed (fig.5).

The first advantage of the EEC nozzles is that the thrust level of existing systems can be uprated with only minor modifications, resulting in increasing range or total impulse for vehicles with length and diameter limitations. In order to limit inert mass, however, these nozzles have to be as light as possible, and are consequently fragile.

Development testing of the EEC nozzles must be conducted at simulated altitude since test article environmental requirements cannot be met at sea level.

Test facility requirements are:

1 — Transient free altitude environment,
2 — Capability for extending the nozzle on the plume,
3 — Plume capture with the extendible nozzle stowed or in case of failure,
4 — Minimum diffuser blowback at motor shutdown,
5 — High accuracy vacuum thrust measurement,
6 — The capability for thrust vector control,
7 — The ability to maintain altitude post-fire until the nozzle has cooled.

Performance restitution

The rocket motor performance can be reduced to one parameter, the vacuum specific impulse (I_{SV}). This parameter is defined as the ratio between total vacuum impulse (I_{FV}) and propellant mass (M) burnt during firing.

$$I_{SV} = \frac{I_{FV}}{M} \quad \text{(m/s)} \tag{a}$$

or:

$$I_{SV} = \frac{I_{FV}}{Mg} \quad \text{(s)} \tag{b}$$

where g is the acceleration of gravity.

In the following, I_{SV} will be defined by (b). I_{FV} is evaluated during the motor burning time, which must be defined exactly. In the same way, the propellant mass M has to be defined precisely: for example, in case of solid rocket motors, two definitions can be held; it can be the propellant mass measured by the manufacturer, but it can also be the total expended mass during firing, assuming that every ejected particle participates in the propulsion.

I_{FV} permits computation of vehicle ΔV, which defines the trajectory of a satellite or the range of a missile:

$$\Delta V = \frac{1}{(M_I - M_F)} \quad I_{FV} \,\mathrm{Log}\,(M_I/M_F)$$

Where: M_I is the mass before firing,
M_F is the mass after firing.

I_{SV} is a characteristic of the motor and allows comparison with other motors.

Simulated altitude tests allow accurate measurement of vacuum thrust with very few corrections. For instance, the correction due to ambient pressure is about 0.5% of the measured thrust.

In conclusion, advantages of captive ground testing at simulated altitude can be summarized as follows:

— performance obtained with minor vacuum correction (with regard to sea level) and with higher accuracy (with regard to flight),

— clean thrust termination,

— measured expended mass very near to flight expended mass (solid rocket motors),

— thermal evaluation of motor, for space operations,

— more instrumentation than flight,

— a test article to examine, after firing,

— cheaper than flight.

1.1.2 Principles

Self-pumping diffuser (supersonic ejector)

The purpose of such a diffuser is to maintain the existing supersonic flow at nozzle exit to the cylindrical wall of a diffuser, where the gas plume impingement takes place. In this way, an ambient static pressure lower than the nozzle exit static pressure can be reached around the base of the motor; this is the simulated altitude pressure.

The diffuser then behaves as an ejector, the expelled rocket gases acting as the pumping fluid of the ejector system. These gases are compressed from the low static pressure just behind the impingement to the static ambient pressure, at the diffuser exit (P_{DE}).

In order to start the diffuser, several geometrical conditions are required for its design. It is necessary to maintain the diffuser exit static pressure lower than a limit starting value ($P_{DE\ MAX}$), which is the result of flow equation relations between nozzle exit and diffuser exit.

Briefly, the governing parameters which determine the diffuser systems are:

— diffuser area to nozzle exit area ratio,
— specific heat ratio of motor exhaust gases,
— nozzle exit Mach number,
— diffuser length to diameter ratio,
— motor chamber pressure (P_C),
— pressure downstream of the diffuser.

(See Section 1.1.3.: empirical rules of ejector design and operation.)

Simulated altitude increases with the diffuser diameter, but the starting pressure ($P_{DE\ MAX}$) decreases at the same time. Above a critical value of simulated altitude, $P_{DE\ MAX}$ is lower than pressure at sea-level. So, it becomes necessary to connect other means of gas extraction downstream to the diffuser.

Supersonic ejector + gas extraction

In most cases, complementary exhausters are connected to the supersonic ejector in order to extract combustion gases. These exhausters can be ejectors (one or several stages connected in parallel or series) or mechanical exhausters.

The capability to maintain low static pressure at the diffuser exit allows the testing of a wide range of motors at different simulated altitudes.

The duration of pumping is a very important characteristic of the test facility, above all for long burning motors such as liquid or cryogenic rocket motors.

Before extraction, gases must be cooled. Water is used in most cases. In this way, it is also possible to eliminate abrasive particles ($Al_2\ 0_3$) or toxic compounds (acids) by keeping them inside a scrubber.

Direct gas extraction

When the gases' flow rate is low (lower than 1 Kg/s for instance) it is possible to extract them directly by means of exhausters.

To protect the exhausters, and in order to improve their efficiency, it is necessary to cool and wash gases before extraction.

This method is also possible in test stands equipped with a diffuser, which works as a duct in this case. The simulated altitude is then defined by the exhauster extraction capability for the given gas flow rate.

Vacuum rooms

These are large test cells in which pre-fire vacuum is provided by pumps. They are only available for very low flow rate of very short burning duration motors. If the flow rate is low, but the burning duration long, pumps can be activated during the test.

Test facilities, equipped with isolation valves, can be also used as vacuum rooms.

1.1.3 Empirical rules of ejector design and operation

Ejector design

Main ejector — diffuser requirements:

Nowaday, the ejectors are designed to serve the dual purpose of evacuating the test cell and performing as supersonic diffusers in propulsion system test installations. Therefore, they must meet the two following basic requirements:

(a) low cell pressure (P_A: ambient)
the ejector — diffuser must be designed to create the simulated altitude pressure in the cell. This cell pressure must be independent from exhaust pressure. It is measured in the area around the expanding jet.

(b) Exhaust pressure (P_{DE}: diffuser exit)

The exhaust pressure (P_{DE}) must be as high as possible in order to extract the flow up to the atmospheric pressure.

There are two basic types of diffuser shapes: cylindrical and with a second throat (fig.6). The second type of diffuser offers a better recompression factor than the first one. Therefore, it is often used in altitude simulation test facility design.

Generally speaking, P_A is obtained through the design of the duct entrance diameter, assuming the duct is properly located (see further). P_{DE} mainly depends on the duct entrance diameter, the contraction ratio and the diffuser length (or second throat length).

Because the very complex nature of the flow in an ejector makes a theoretical design infeasible, semi-empirical methods have been developed for designing ejectors for various applications.

Let us describe the main parameters which the designer has to take into account (fig.7).

Ejector diffuser geometry:

(a) D_D; A_D: duct entrance diameter and area,

(b) L: nozzle exit to ramp section distance,

(c) θ_r: ramp section angle,

(d) D_{ST}; A_{ST}: second throat diameter and area,

(e) L_{ST}: second throat length,

(f) θ_{ST}: subsonic diffuser angle,

(g) D_{SD}; A_{SD}: subsonic diffuser exit diameter and area.

We must add to the list one of the most important factors which must be considered during the design of a second throat diffuser. This is the contraction ratio defined by:

$\psi = (D_{ST}/D_D)^2$

Relations Nozzle Exit/Duct entrance.

(h) σ: expansion coefficient of free jet:
$\sigma = (D_D/D_{NE})^2 = A_D/A_{NE}$
(D_{NE}: diameter of nozzle exit).

The most important parameters are: ψ, D_D, L, L_{LST}.

REMARK: In this list, it is supposed that the ejector-diffuser has a second throat. Effectively, this configuration increases the ejector starting and operating ratios. The improvement increases as the contraction ratio is decreased. But, if the performance of the downstream exhausters is good enough, a cylindrical ejector system is generally sufficient. The design, maintenance and replacement aspects are simpler in any case.

Design:

(a) D_d, duct entrance diameter:
Starting P_C/P_{DE} [$(P_C/P_{DE})_{ST}$] increases linearly with (A_D/A^*) when this last ratio varies between 1 and 80; beyond 80, the increase is more important. The larger D_d is, the lower the cell pressure is (to a reasonable extent) (see 1.1.2. and further). But when the free jet expansion ratio $\sigma = \frac{A_D}{A_{NE}}$ is greater than 5, ejector starting may be impossible or at least difficult. The greater σ is, the greater the Mach at the re-attachment of the supersonic jet: the loss of total pressure may be then too great to allow the flow to pass through the duct (especially the second throat). Moreover, the greater σ is, the larger the second throat diameter must be. So, it is recommended to design D_D in order to achieve:

$$\begin{cases} -\ \sigma = A_D/A_{NE} < 5 \\ -\ \text{ratio } P_A/P_C \text{ according to simulated altitude requirement } (P_A). \end{cases}$$

(b) L, nozzle exit to ramp section distance:
If the second throat is located too far downstream, the free jet impinges upon the ramp and the cell pressure increases. If L is too great, the ejector may not start: the decrease of Mach number, due to friction and re-compression through oblique shock wave along the duct entrance, may be too significant to allow full flow (supersonic) in the second throat. Therefore, L must be designed so that the free jet impingement will be located at a reasonable distance from the ramp.

In any case, it is wise to calculate the free jet expansion and impingement in order to determine L. Usually, it is known that L/D_D may vary between 0,8 and 2, mainly depending on $\frac{(D_D)^2}{D^*}$. The best value seems to be $L/D_D = 1$.

Another criterion of choice may appear with EEC nozzles. The designer will have to verify that the plume does not impinge on the ramp section after deployment.

(c) θ_r, ramp section angle:
It is not an important parameter, if it is lower than 15°. Good results are obtained with $\theta_r = 6°$.

(d) D_{ST}, second throat diameter:
It is recommended to design D_{ST} slightly greater than D_{NE}. This condition is obviously achieved with a cylindrical diffuser.

D_D and D_{ST} must be designed in order to obtain an acceptable value for the contraction ratio $\psi = \frac{A_{ST}}{A_D}$.

The normal shock starting limit gives a minimum which is always higher than the experimental one. But it is often hazardous to design ψ at the experimental minimum value.

It is therefore recommended to choose for ψ a value between the theoretical value calculated with the well-known normal shock method and the following semi-empirical value which generally gives good results (fig.8).

$$\boxed{\psi_{min}^{(SE)} = \left\{\frac{1}{\Lambda}\left[(0.74\,\gamma - 0.18)(\Lambda - 5.62) + 6.05\right]\right\}^2}$$

where: $\Lambda = \frac{A_D}{A^*}$

$\gamma = \frac{C_p}{C_v}$ of combustion gas.

(e) L_{ST}, second throat length:
In order to obtain a good recompression through the diffuser, the second throat must be long enough to shelter the shock system which is responsible for the recompression. The efficiency of the ejector diffuser improves when L_{ST}/D_{ST} increases from 0 to 8 and then remains constant. For altitude simulation application, a long second throat ejector diffuser is recommended ($L_{ST}/D_{ST} = 8$). This is also available for cylindrical ejector diffuser.

(f) Summary (table 1):

$$\sigma = \frac{A_D}{A_{NE}} \leqslant 5$$

$$L/D_D = 1$$

$$\theta_r = 6°$$

$$\psi \geqslant \psi_{min}^{(sc)}$$

$$A_{ST} \geqslant A_{NE}$$

$$L_{ST}/D_{ST} = 8$$

Ejector starting and operation

P_{DE} remaining constant, and P_c increasing, three cases successively occur:

(a) Flow separation in nozzle (fig.9a)
When P_C is about twice P_{DE}, the flow in the nozzle becomes sonic and undergoes a recompression in the nozzle cone through a shock-wave whose location is determined by the P_C value.

(b) Full-flow in nozzle (fig.9b)
P_C increasing, the shock wave leaves the nozzle. Downstream the nozzle, the flow is supersonic then it is recompressed into a subsonic flow along the diffuser. This phenomenon is very unstable and noisy.

(c) Full-flow in ejector diffuser system (fig.9c)
If P_C increases again, the supersonic jet expands until it reaches the diffuser wall. It is then recompressed through an oblic shock waves system whose location is independent of the P_C value. One says that the diffuser has "started", which can be also defined as the attachment of the shock wave at the diffuser wall.

Fig.10 shows the variation of P_A/P_{DE} ratio versus P_C/P_{DE}. It can be seen that P_A/P_{DE} reaches a minimum value which is only slightly affected by P_C/P_{DE} increases (increase of P_C or reduction of P_{DE}) and which characterizes the diffuser starting and operating.

Theoretical methods are available for predicting the starting presure ratio for a zero secondary flow ejector system having a second throat diffuser. The one-dimensional conservation equations are used (Constant-pressure mixing theory, constant area mixing theory,....). These methods take into account the wall friction along the tube and the thermal losses. But these methods are very complicated and require the use of a high speed computer since the cell pressure and the ramp static pressure evolution must be known prior to the calculation. But empirical design methods have been developed from experimental results and are easily employed for practical applications.

In practice, ejector starting and operation performance are approximately 80 to 90 percent of the theoretical normal shock value $(P_{t2}/P_{t1})_{ns}$ calculated at the duct entrance, that is to say:

$$(P_a/P_t) \simeq 0{,}8 \text{ or } 0{,}9 \times (P_{t2}/P_{t1})_{ns}$$

The numerical coefficient depends on the nozzle geometry, the diffuser length-to-diameter ratio.

The total pressure recovery must be evaluated across a normal shock for the one-dimensional isentropic Mach number corresponding to the area ratio of the duct entrance in the region of jet impingement to the nozzle throat area.

1.1.4 Means of providing vacuum

There are many ways to provide vacuum, depending on the mass flow rates which must be exhausted. But two families have to be considered:

— ejectors,
— mechanical exhausters.

Ejectors

These operate using a primary flow of:

— steam, or
— boiled water, or
— oil, or
— nitrogen or air.

Ejectors using steam or boiled water are the most efficient because of their high momentum. They are used both for gas extraction downstream of a diffuser and for providing vacuum inside the cell before ignition, rather quickly.

Oil ejectors allow very high simulated altitudes to be reached before ignition, but in a rather long time. They are not used to extract gases. Nitrogen or air ejectors are used in centres which already use these gases to operate other facilities (such as air-breathing facilities. See AGARDograph No.269). Nitrogen ejectors are sometimes used in the same way as oil ejectors. Air ejectors must not be used for direct combustion gas extraction, because of post-combustion hazards. Air ejectors are therefore set downstream of steam ejectors, as second stages or more.

Ejectors, because they need great amounts of primary flows to operate, are generally limited in duration.

Nevertheless, there are few difficulties for maintenance with them and they do not risk pollution by gases.

Mechanical exhausters (rotary machinery)

Small mechanical pumps are used either to provide vacuum before ignition, or to maintain it during firing of low flow rates motors. Big mechanical pumps, for gas extraction, have the great advantage of long operating duration, but they need enormous amounts of power and maintenance is rather heavy. In addition, they can fail during combustion of the tested motor, so they must be either reliable or redundant. Finally, they are generally more expensive than ejector systems, which is why they are not used in recently established facilities.

Conclusion

Steam or hot water ejectors are the most often used, because of low costs, easy maintenance and high efficiency. But they are limited in operating duration.

The different ways of producing steam or hot water are described in the following paragraph.

1.1.5 Steam or hot water generation

A distinction has to be made, concerning steam generation, between solid propellant and liquid (or cryogenic) rocket motor tests. In the last case, combustion can be stopped by closing the propellant injection valves. In the first case, it is absolutely impossible to stop combustion before the end of burning. Therefore, the steam generation system used for testing solid propellant motors must have a high level of reliability.

On the other hand, solid propellant motors have short duration combustions, compared to liquid motors.

Briefly, steam generation systems must ensure:

— reliability with regard to safety questions, for solid propellant motor tests,
— a long operating duration of ejectors, for liquid motor tests.

Consequently, two ways of producing steam are generally considered:

(a) Steam is produced by a boiler (fuel fixed boiler; flash boiler; etc...) and stored in accumulators. Steam is then over-saturated, at about 250°C and 4.5 MPa. When consumed, steam is automatically re-generated by hot water stored in the accumulator. Water to be converted into steam must be previously demineralized. Therefore, steam is very clean and there is no risk of polluting the ejectors. Nevertheless, several hours are necessary to charge the accumulator prior to the test, and the consumption of steam is of course limited to the total amount stored in the accumulator. Thus, the use of such a system must be correctly planned in advance. Though the accumulator represents a serious investment of the beginning, it needs few maintenance operations. Moreover, it is very reliable in operation, so it is often used for performing solid propellant motor tests.

(b) Steam is produced by means of a combustor generator and consumed directly ("real time"). Water under pressure is injected into, and mixed with, the high temperature gases issued from the combustor generator. These combustor generators are often recuperated liquid rocket motors which have been adapted to this special use, but some specific systems are now available. The propellants are generally either oil (fuel) and Nitric acid (oxidiser), or alcohol (fuel) and Lox (oxidiser). Therefore, it must be noted that steam is contaminated with the exhaust gases. The resulting installations based on combustor generators are smaller than steam accumulators, and properly suited to intermittent operations. Though the initial investment seems to be cheaper than accumulators, this advantage may be reduced by costly maintenance.

1.1.6 Performance

In this paragraph, we try to take into account a synthesis of the parameters, which are the most interesting ones, before performing a rocket motor test. We don't take into account any financial and security consideration. This paragraph will be considered as a reference when establishing the characteristics of Altitude test cells, in part 3 (§ 3.2).

1.1.6.1 *General performance*

The following parameters allow a quick understanding of the facility capability.

— simulated altitude level: unit: Km, or feet
The standard ambient pressure, at this altitude can be precised.

— Type of propellant: test facilities are often specific either to solid propellant motors, or to liquid motors (safety questions; propellant storage, etc...).

— Firing orientation: horizontal, vertically down.

— Cell chamber dimensions: unit: metres, or feet.

— Access: type (hatch; cover top door; clamshell door; etc...) and dimensions (metres or feet).
If cranes, or so on, are available, this should be indicated here.

— Max thrust or max flow rate: unit: KN or lbf; Kg/s or lbm/s.

— Max firing duration: unit: seconds,
If test duration capability depends on the motor mass flow rate, it must be specified here.

1.1.6.2 *Complementary description*

The following parameters allow a good understanding of how the facility operates. Here must also be pointed out the specific device, temperature conditioning, etc...

— Diffuser-ejector: type (cylindrical, second thrust, centre-body), diameter of diffuser entrance and pressure recovery ratio.

— Exhausters: type (steam ejector; mechanical) power or steam supply and typical (or maximum) exhausted mass flow-rate versus exhaust pressure.

— Specific devices: for example:
- spinning device: brief description and performance (rpm).
- temperature conditioning: environmental temperature, propellant conditioning.
- special safety device (nitrogen purge)
- Isolation valve.
- Annular ejector.
- Thrust control vector capability.

1.1.6.3 *Controls and data*

This is a brief description of the capability to provide pre-fire electrical controls and calibration (example: thrust), then data acquisition and processing. It is also necessary to provide information about photography capability (TV; high velocity movie cameras), and any device which can improve the efficiency of measurements.

1.1.6.4 *Typical accuracies*

According to one of the main requirements (accurate vacuum specific impulse, I_{SV}, measurement), the accuracy on the main parameters measurements which are used for I_{SV} and ΔI_{SV} (uncertainty) calculations have to be specified: thrust, combustion pressure, ambient (vacuum) pressure, nozzle exit area, mass flow rate (liquid), expanded mass (solid) and, finally I_{sv}.

Accuracy is defined as a percentage of full scale of measurement, so that the typical range must be specified:

Example: Thrust: 0.2%; 100 000 lbf.

1.2 SPECIFIC QUESTIONS

1.2.1 Safety

No distinction is made between civilian and military motors. Both of them are tested in similar conditions. The discrimination relates to different propellants.

Extreme care must be taken about safety questions. Each country follows safety rules which have the strength of law.

In any case, before testing or after any accident, a special safety enquiry is absolutely necessary, to know exactly what to do, how to do it, and why it is done.

To spare human lives must be the first concern of any people working in rocket-motor test centres.

Generally, hazards are the same as on sea-level ground test-stands. During simulated altitude tests, these hazards can be aggravated when wrong operation of the facility occurs, or because of its limited volume.

The basic aggravating factors are:

(1) The ejector-diffuser operates badly (no starting or failing): this is generally due to ill adaptation of the diffuser to the rocket engine (see § 1.1.3.). That may also occur when the impingement zone is too much eroded, decreasing the σ ratio below the threshold (particularly, when ablative thermal coat is used). Finally, the wall of diffuser can collapse, letting pressurized cooling water (see § 1.2.5) enter the duct and considerably cool and slow down the gas flow, which leads to ejector failure. Therefore, a large part of the flow is forced back to the rear of the motor, which can be seriously damaged. That can lead to explosion and fire. In fact, very hot gases burn and slash fibres of solid propellant motor composite cases or liquid propellant distribution pipes. Then, the high pressurized motor case explodes and the remaining solid propellant may burn on; that can bring about a test cell explosion. As for liquid propellants, leakages bring about severe fires and eventually test cell demolition.

(2) Post-combustion of gases issued from combustion, or leakages of propellants.

Gases from combustion contain high reducing agents (such as hydrogen). If no care is taken, they are oxidised by the

oxygen present in the air, inside the facility. That may lead to the demolition of the facility, because of explosions. Another hazard, due to post combustion must be taken into account because of its possible effects on human lives: at the end of the test fire, a large part of the facility may be full of those reducing gases. When opening the test cell in order to enter it and to have a look at the motor, the recompression to atmospheric pressure brings a lot of oxygen and may induce an explosive chemical reaction. This may happen with leakages of both oxidisers or fuels. For instance, there are great hazards with oxygen hydrogen leakages: oxygen is heavy compared to air and burns as long as its concentration in air is higher than normal; hydrogen violently explodes as soon as its concentration in the air has passed beyond the explosiveness threshold.

(3) The limited volume:
When propellant burns, after an accident for example, the limited volume offers few emergency exits and fire quickly increases pressure, so it may explode. Alternatively, people who are inside the facility may be killed by high heat flux, even if they are protected against shrapnel and flames. In the case of leakage, the inflammability or explosiveness thresholds are reached very quickly within such a limited volume. Finally, the confined atmosphere of a test cell increases the risk of inhaling toxic liquid propellant vapours.

(4) Rigid walls in the immediate vicinity of the rocket motor:
When testing a high energy propellant (HMX, X-Link double base) rocket-motor, an initial explosion may turn to detonation. Because of explosion, grain is cut into pieces which are flung violently against the cell walls. Then detonation may occur, because of shock to detonation transition (SDT) phenomenon. The first detonation is transmitted to over grain pieces, in a chain reaction SDT type. Finally, the detonation of all the remaining grain is highly probable. As a result, the test cell and certainly the whole facility are destroyed; moreover, shock waves and blown splinters are quite dangerous for the neighbourhood.

Therefore, great care has to be taken:

— Technological care:
It is good to be conservative with regard to design and operation rules for ejectors. A safe operation is better than hazardous performance. As for the cooling technology of ejector-diffuser system, it must be perfectly known and mastered (ablative thermal coat; film cooling: water jacketed tube). In order to prevent test cell explosion when propellant burns inside it, it is wise to plan a safety valve, which can also be designed (and used) as an access to the cell.

— To respect safety rules:
Safety distances between buildings, number and ordering of emergency exits, storage rules, handling procedures have been stated from experience and are always improved in order to protect people in regard to pyrotechnic, hypergolic and cryogenic hazards. With high energy solid propellants (nitro-binder composite propellants), those rules must be more and more strictly complied with, because of the risk of detonation.

— Concerning post combustion and propellants vapours leakages, it is wise to systematically sweep the facility with nitrogen in order to neutralize the inside atmosphere. As for cryogenic motor test facilities, a permanent control of both oxygen and hydrogen concentrations must be provided. Even during firing test, a nitrogen purge inside the test cell is necessary to avoid the formation of explosive vapour pockets. The recompression of the test cell to atmospheric pressure at the end of firing test, must be provided with nitrogen before opening it. Then, entering the test cell is allowed only when the atmosphere is breathable.

— High energy propellant detonation: some experts maintain it is advantageous to set panels of dumping materials between the rocket motor and the cell chamber walls. In that case, grain pieces lose a significant part of their energy, when going through the panels. Thus, their total energy is not high enough to allow SDT (Shock to Detonation Transition) when colliding with the walls. Nevertheless, the bigger the grain pieces are, the lower the efficiency of dumping panels is: after explosion, big pieces of motor grain are likely to go through the panels without significant energy losses, so SDT remains possible.

1.2.2 Accuracy

This part of the work has already been undertaken within the frame of PEP/WG17 activities. For more information, read AGARD Report No.230. Briefly, it is reasonable to consider that a rather good estimation of the uncertainty on I_{SV}, by measurement and data processing, is about 0,5%, for values of I_{SV} about 300 s.

A difficulty remains in the lack of standard methodology for evaluating accuracy. Each test centre can, of course, have its own methodology, but, in order to permit easier comparisons between motor performances, a standard methodology should be proposed (by AGARD, for instance).

1.2.3 Transient phases

Transient phases may be as delicate for fragile nozzle behaviour as for cinematography. Variations of ambient pressure (P_A) induced during transient phases may either damage the nozzle, or flow back hot gases and dirty particles.

The main transient phases to take into consideration are:

1.2.3.1 *Ignition, then supersonic ejector starting*

- The main requirement is to obtain an ambient pressure before ignition (P_A) close to the one after the supersonic ejector starting (P_A^E; E for equilibrium).

- If P_A^O is significantly lower than expected P_A^E, hot and dirty particles will flow back before equilibrium, which may damage rear nozzle equipment (such as actuators) and obscure camera lenses.

1.2.3.2 *EEC extension*

During extension, gases may flow laterally through slots between rings or panels. This flow is quantitatively negligible compared to the rocket exhaust gas flow. Provided that the volume surrounding the rocket is significant (several tens of cubic metres), the consecutive cell pressure increase will be quite invisible. The only thing to ensure is that the ejector will start in both configurations.

1.2.3.3 *Thrust vector control* (Fig.11)

Thrust vector control is today obtained by making the nozzle rotate around its axis, such as pitch or yaw, by means of actuators.

This rotation leads to asymmetry of the plume compared to the ejector.

Provided that the impingement remains on the ejector wall, the slight rotation of nozzle (a few degrees) modifies the impingement angle (fig.) slightly.

Because $\rho_i^r + \rho_i^o$, P_{iu} + is greater than ρ_{iu}^o (Pressure upstream Impingement) and because $\rho_i^r - < \rho_i^o$, $P_{i\bar{u}} < P_{iu}^o$.

Thus, a slight transverse flow will be generated in the rear motor area, but without great modification of ambient pressure (P_A), and without any consequence on ejector operation.

1.2.3.4 *End of burning*

When motor propellant stops burning, gas momentum will quickly decrease so that supersonic flow is impossible inside the ejector-diffuser system.

Thus, the higher pressure at diffuser exit (P_{DE}) compared to the cell chamber (P_A) will force the remaining gas to flow back towards the motor.

If nothing is done to reduce this phenomenon, several hundreds of millibars (1 millibar: 10^2 P_A — about 0.015 psi) superpressure will be violently applied to the nozzle, blowing dirt particles on the cell chamber walls, camera lenses, and so on.

Even when using specific anti-back-flow devices (see § 1.2.4), it is quite impossible to eliminate this phenomenon totally.

1.2.4 Anti back-flow devices

There are generally two ways of reducing back-flow phenomenon:

— Downstream diffuser exit auxiliary ejector. At the end of burning, this ejector is started, in order to replace the ejector-diffuser system. So the combustion gas continue to flow inside the diffuser at a supersonic rate.

It is generally a steam ejector, set downstream of the diffuser exit, either on the axis, or tangential to the wall (annular ejector).

Nevertheless, it is necessary to be sure that the steam (or other fluid) issued from auxiliary ejector will not jam the combustion gas flow, instead of sucking it downstream.

It is generally good to adjust the pressure of feed steam so that the ultimate zero flow sucking pressure is close to the ambient cell pressure previous to end burning.

— Quick-closing shutter valve, which is set at the diffuser exit. This valve is normally actuated by an explosive charge or by gaseous nitrogen. But shutters are designed (they are asymmetric in regard to the duct flow) in order to immediately close if the flow reverses direction towards the rocket motor.

Because such a valve is not leaktight, it is necessary to use another valve, specially designed to ensure this function.

2 LIST AND TEST CAPACITIES OF ORGANISATIONS

2.1 LIST OF ORGANISATIONS

Country	Organisation/location	Test facilities included in this register			Page
		SP	LP	CP	
FRANCE	CAEPE: Centre d'Achèvement et d'Essais de Propulseurs et Engins/Saint Médard en Jalles	1			12
	ONERA: Office National d'Etudes et Recherches Aérospatiales/Chalais-Meudon	2			12
	Modane		(2)		
	Palaiseau		(2)		
	SEP: Société Européenne de Propulsion/Vernon			1	12
GERMANY	DFVLR: Deutsche Forschungs-und Versuchsanstalt für Luft und Raumfahrt/ Lampoldshausen		3		13
ITALY	SNIA-BPD/Colleferro- Roma	2			
UNITED KINGDOM	RAE: Royal Aircraft Establishment/Pyestock	1			13
	RO: Royal Ordnance/Westcott	1	1		14
UNITED STATES of AMERICA	AEDC-AFS: Arnold Enginering and Development Center — Air Force Station/Tullahoma (TN)	2			14
			(2)		
	AFRPL: Air Force Rocket Propulsion Laboratory/Edwards Air Force Base (CA)	2	5		15
	ATC: Aerojet Techsystems Company/Sacramento (CA)		(2)		15
	MJL: Marquardt Jet Laboratory The Marquardt Company/Van Nuys (CA)		7		
	WSTF — NASA: White Sands Test Facility — National Aeronautics and Space Administration/ Las Cruces (NM)	1	1		16
		(1)			
			(1)		

Numbers in brackets refer to test cells available for different types of propellants.
SP = Solid propellant; LP = Liquid propellant; CP = Cryogenic propellant.

2.2 TEST CAPACITIES OF ORGANISATION

1 Organisation

Name: Centre d'Achèvement et d'Essais de Propulseurs et Engins

Address: BP 2
33 165 — Saint Médard en Jalles Cédex
FRANCE
Tel. (33) 56 05 84 85

Contact: Monsieur le Directeur

2 Test Cells

Designation	Propellant	Simulated altitude	Max thrust or Max flow rate
MESA	Solid	27 km (90 000 ft)	600 kN (135 000 lbf)

1 Organisation

Name: Office National d'Etudes et Recherches Aérospatiales

Address: 29, Avenue de la Division Leclerc
92 322 — Châtillon Cedex
FRANCE
Tel. (33) (1) 46 57 11 60

Contact: Monsieur le Directeur

2 Test Cells

Designation	Propellant	Simulated altitude	Max thrust or Max flow rate
R2 CH	Solid	35 km (120 000 ft)	0.5 kN (115 lbf)
R3 CH	Solid	50 km (170 000 ft)	0.5 kN (115 lbf)
A 75	Solid/Liquid	80 km (270 000 ft)	
A 611	Solid/Liquid	80 km (270 000 ft)	
S4 MA	Solid/Liquid	25 km (85 000 ft)	
S4 B	Solid/Liquid	25 km (85 000 ft)	

1 Organisation

Name: Société Européenne de Propulsion

Address: BP 802
27 207 — Vernon
FRANCE
Tel. (33) 32 51 31 21

Contact: Monsieur le Directeur

2 Test Cells

Designation	Propellant	Simulated altitude	Max thrust or Max flow rate
PF 41	Cryogenic (LH2/LOX)	25 km (85 000 ft)	70 kN (16 000 lbf)

1 Organisation

Name: Deutsche Forschungs und Versuchsanstalt für Luft und Raumfahrt
Forschungsbereich Energetik
Institut für Chemische Antriebe und Verfahrenstechnik

Address: D. 7101 Hardthausen Am Kocher
GERMANY

Contact: Professor Lo, Director

2 Test Cells

Designation	Propellant	Simulated altitude	Max thrust or Max flow rate
P 1.5	Liquid	35 km (115 000 ft)	0.5 kN (110 lbf)
P 3	Liquid	24 km (80 000 ft)	70 kN (16 000 lbf)
P 4.2	Liquid	24 km (80 000 ft)	800 kN (180 000 lbf)

1 Organisation

Name: SNIA — BPD

Address: Corso Garibaldi, 22
00034 Colleferro (ROMA)
ITALY

Contact: Mr Conte, Head Engineering Department

2 Test Cells

Designation	Propellant	Simulated altitude	Max thrust or Max flow rate
ISA 1	Solid	27 km (90 000 ft)	22 kN (5 000 lbf)
ISA 2	Solid	27 km (90 000 ft)	50 kN (11 500 lbf)

1 Organisation

Name: Royal Aircraft Establishment

Address: Pyestock — Farnborough Hants
GU14 0LS
UNITED KINGDOM
Tel. (0252) 54 44 11

Contact:

2 Test Cells

Designation	Propellant	Simulated altitude	Max thrust or Max flow rate
Cell 2	Solid	30 km (100 000 ft)	30 kN (7 000 lbf)

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1 **Organisation**

Name: Royal Ordnance
Explosives Division
Research and Development Centre

Address: Westcott — Aylesbury Buckinghamshire
HP18 0NZ
UNITED KINGDOM
Tel. (0296) 65 11 11

Contact: Mr Whitehouse

2 **Test Cells**

Designation	Propellant	Simulated altitude	Max thrust or Max flow rate
Site 6/Cell 1	Solid	35 km (115 000 ft)	49 kN (11 000 lbf)
Site 6/Cell 2	Liquid	35 km (115 000 ft)	0.01 kg/s (0.023 lbm/s)

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1 **Organisation**

Name: Arnold Engineering and Development Center
Arnold Air Force Station

Address: Tennessee 37389
UNITED STATES OF AMERICA
Tel. (615) 455 26 11
454 77 55

Contact: Mr Carlton Garner
AEDC/DOCA

2 **Test Cells**

Designation	Propellant	Simulated altitude	Max thrust or Max flow rate
T 3	Solid	52 km (170 000 ft)	89 kN (20 000 lbf)
J 3	Solid/Liquid	38 km (125 000 ft)	890 kN (200 000 lbf)
J 4	Solid/Liquid	30 km (100 000 ft)	2224 kN (500 000 lbf)
J 5	Solid/Liquid	46 km (150 000 ft)	556 kN (125 000 lbf)

1 Organisation

Name: Air Force Rocket Propulsion Laboratory
Edwards Air Force Base

Address: California 93535
UNITED STATES OF AMERICA
Tel. (805) 277 56 20

Contact: Col. Ross Nunn, Commander

2 Test Cells

Designation	Propellant	Simulated altitude	Max thrust or Max flow rate
Area 1.42/A	Solid	33 km (110 000 ft)	220 kN (50 000 lbf)
Area 1.42/B	Liquid	33 km (110 000 ft)	220 kN (50 000 lbf)
Area 1.42/D	Solid	33 km (110 000 ft)	90 kN (20 000 lbf)
Area 1.14/A	Liquid	30 km (100 000 ft)	22 kN (5 000 lbf)
Area 1.14/C	Liquid	40 km (130 000 ft)	0.4 kN (90 lbf)
Area 1.14/D	Liquid	40 km (130 000ft)	6.6 kN (1 500 lbf)
Area 1.14/E	Liquid	40 km (130 000 ft)	22 kN (5 000 lbf)

1 Organisation

Name: Aerojet Techsystems Company

Address: PO Box 13 222 — Sacramento, California 95813
UNITED STATES OF AMERICA
Tel. (916) 355 35 20

Contact: Mr Brincka, Director, Test Operations

2 Test Cells

Designation	Propellant	Simulated altitude	Max thrust or Max flow rate
J 3	Solid/Liquid	57 km (190 000 ft)	7 kN (1 500 lbf)
J 4	Solid/Liquid	30 km (100 000 ft)	90 kN (20 000 lbf)

1 Organisation

Name: Marquardt Jet Laboratory
The Marquardt Company

Address: 16 555 Saticoy Street — Van Nuys, California 91409
UNITED STATES OF AMERICA
Tel. (818) 989 64 00

Contact: Mr Sund, Director Engineering

2 Test Cells

Designation	Propellant	Simulated altitude	Max thrust or Max flow rate
Cell 1	Liquid	40 km (130 000 ft)	5.5 kN (1 200 lbf)
Cell 2 Cell 8	Liquid	40 km (130 000 ft)	180 kN (40 000 lbf)
Cell 9 #1	Liquid	71 km (235 000 ft)	0.05 kN (11 lbf)
Cell 9 #2	Liquid	71 km (235 000 ft)	5 kN (1 100 lbf)
PRL Pads D and E	Liquid	40 km (130 000 ft)	0.5 kN (110 lbf)

1 Organisation

Name: White Sands Test Facility
National Aeronautics and Space Administration

Address: PO Drawer MM — Las Cruces, New Mexico 88004
UNITED STATES OF AMERICA
Tel. (505) 524 50 11

Contact: Mr Tillett, Manager

2 Test Cells

Designation	Propellant	Simulated altitude	Max thrust or Max flow rate
TS 302	Solid/Liquid	75 km (250 000 ft)	
TS 401	Liquid	35 km (120 000 ft)	100 kN (22 500 lbf)
TS 403	Liquid	35 km (120 000 ft)	100 kN (22 500 lbf)
TS 405	Solid	30 km (100 000 ft)	90 kN (20 000 lbf)

3 ALTITUDE TEST FACILITIES

3.1 LIST OF TEST FACILITIES

Propellant SP	LP	CP	Altitude km	Max Thrust kN	Test Cell Designation	Page	Organisation Name	Page
*			27	600	MESA	19	CAEPE	12
*			35	0.5	R2 CH	21	ONERA	12
*			50	0.5	R3 CH	22	ONERA	
*	*		80		A 75	23	ONERA	
*	*		80		A 611	23	ONERA	
*	*		25		S4 MA	24	ONERA	
*	*		25		S4 B	25	ONERA	
		*	25	70	PF 41	26	SEP	12
	*		35	5.5	P 1.5	27	DFVLR	
	*		24	70	P 3	28	DFVLR	
	*		24	800	P 4.2	29	DFVLR	
*			27	22	ISA 1	30	SNIA.BPD	13
*			27	50	ISA 2	31	SNIA.BPD	
*			30	30	Cell 2	32	RAE	13
*			35	49	Cell 1	33	RO	14
	*		35		Cell 2	33	RO	
*			59	90	T 3	34	AEDC	14
*	*		30	450	J 3	36	AEDC	
*	*		30	2 250	J 4	38	AEDC	
*	*		45	600	J 5	40	AEDC	
*			33	220	1.42/A	42	AFRPL	15
	*		33	220	1 42/B	44	AFRPL	
*			33	90	1.42/D	46	AFRPL	
	*		30	22	1.14/A	48	AFRPL	
	*		40	0.4	1.14/C	49	AFRPL	
	*		40	6.6	1.14/D	50	AFRPL	
	*		40	22	1.14/E	51	AFRPL	
*	*	*	57	7	J 3	52	ATC	15
*	*	*	30	90	J 4	53	ATC	
	*		40	5.5	Cell 1	54	MJL	15
	*		40	180	Cell 2	55	MJL	
	*		40	180	Cell 8	55	MJL	
	*		71	0.05	Cell 9 #1	56	MJL	
	*		71	5	Cell 9 #2	57	MJL	
	*		40	0.5	PRL Pad D	58	MJL	
	*		40	0.5	PRL Pad E	58	MJL	
*	*		75		TS 302	59	WSTF	16
	*		35	100	TS 401	60	WSTF	
	*		35	100	TS 403	61	WSTF	
*			30	90	TS 405	62	WSTF	

3.2 LIST OF TEST FACILITIES WITH SPECIAL CAPABILITIES

Abbreviations:

TC = Temperature Conditioning
ABF = Anti Back-Flow device
S = Spinning
TVC = Thrust Vector Control
IPC = In Place Calibration
MQ = Motor Quench
EGS = Exhaust Gas Scrubber

Organisation	Test Cell Designation	TC	ABF	S	TVC	IPC	MQ	EGS
CAEPE	MESA	*	*	*	*	*		*
SEP	PF 41		*			*		
DFVLR	P 1.5		*			*		
DFVLR	P 3		*			*		
DFVLR	P 4.2					*		
SNIA.BPD	ISA 1		*					
SNIA.BPD	ISA 2		*	*			*	
RAE	Cell 2		*			*		*
AEDC	T 3	*	*		*	*	*	*
AEDC	J 3	*	*		*	*	*	*
AEDC	J 4	*	*		*	*	*	*
AEDC	J 5	*	*	*	*	*	*	*
AFRPL	1.42/A	*		*		*		*
AFRPL	1.42/B	*				*		*
AFRPL	1.42/D	*				*		*
AFRPL	1.14/A	*				*		*
AFRPL	1.14/C	*				*		*
AFRPL	1.14/D	*				*		*
AFRPL	1.14/E	*				*		*
ATC	J 3	*	*			*		*
ATC	J 4	*	*	*	*	*		*
MJL	Cell 1		*		*			*
MJL	Cell 2				*			*
MJL	Cell 8				*			*
MJL	Cell 9 #1	*						*
MJL	Cell 9 #2	*						*
MJL	PRL Pad D	*						*
MJL	PRL Pad E	*						*
WSTF	TS 302	*						
WSTF	TS 401	*	*					
WSTF	TS 403	*	*					
WSTF	TS 405		*	*				

3.3 TEST FACILITIES

ALTITUDE: 30 km

1. Identification

organisation	location	designation
Centre d'Achèvement et d'Essais des Propulseurs et Engins (C.A.E.P.E.)	Saint Médard en Jalles (France)	MESA

2. Contact

Monsieur le Directeur
BP 2
33165-St Médard en Jalles-Cédex

3. General Performance

Type of engine:	Solid propellant rocket motor
Firing orientation:	Horizontal
Test cell size:	Diam. 6 m — Length 12 m
Capacities: mean mass flow:	120 kg/s
max thrust:	600 kN
Max firing duration:	90 s
Transients:	Exit cone extension Thrust vectoring

4. Technical equipment

Temperature conditioning (0°C—45°C)
Spinning device (30 rpm)
Exhaust gas diffuser
Steam ejectors (steam stored in accumulators)
Isolation valve
Annular ejector
Anti back-flow device
Exhaust gas scrubber

5. Data acquisition and processing

Computer controlled data acquisition system (SEMS SOLAR 16—65 and 16—75; 400 000 samples per second) including real time graphics systems, analog recorders and video.

6. Measuring equipment

Thrust	460 kN
Digital acquisition	100 Strain gauge transducers 100 Strain gauges 80 Thermocouples 20 General purpose transducers 25 Timing 6 Optical fibres
Analog acquisition	72 Piezo electric transducers 2 Thrust vector control 12 Ignition lines 5 Video

7. Accuracy

measurement	%	range
thrust. kN (lbf)	0.5	230 — 460 (50 000 — 100 000)
propellant expanded mass Kg (lbm)	0.1	2 300 — 7 300 (5 000 — 16 000)
combustion pressure MPa (psia)	0.15	5 — 10 (750 — 1500)
vacuum pressure KPa (psia)	0.2	0.7 — 21 (0.1 — 3.)
I_{sv} .sec	0.4	#300

ALTITUDE: 35 km

1. Identification

organisation	location	designation
Office National d'Etudes et de Recherches Aérospatiales (ONERA)	Châtillon (France)	Centre de Chalais R 2 CH

2. Contact

Monsieur le Directeur
29 Avenue de la Division Leclerc
92332 Châtillon Cedex

3. General Performance

Type of engine:	Solid propellant reduced scale motor
Firing orientation:	Horizontal
Test cell size:	$1.7 \times 1.05 \times 1.3$ m × m × m (H × l × L)
Capacities: mean mass flow:	
max thrust:	0.5 kN
Max firing duration:	20 s
Transients:	Vehicle staging simulation Exit cone extension simulation

4. Technical equipment

Exhaust: vacuum tank 500 m^3
Exhaust gas diffuser
Mach 3 to Mach 7 flow

5. Data acquisition and processing

ALTITUDE: 50 km

1. Identification

organisation	location	designation
Office National d'Etudes et de Recherches Aérospatiales (ONERA)	Châtillon (France)	Centre de Chalais R 3 CH

2. Contact

Monsieur le Directeur
29 Avenue de la Division Leclerc
92332 Châtillon Cedex

3. General Performance

Type of engine:	Solid propellant reduced scale motor
Firing orientation:	Horizontal
Test cell size:	1.7 × 1.05 × 1.3 m × m × m (H × l × L)
Capacities: mean mass flow:	
max thrust:	0.5 kN
Max firing duration:	10 s
Transients:	Vehicle staging simulation Exit cone extension simulation

4. Technical equipment

Exhaust: vacuum tank 500 m^3
Exhaust gas diffuser
Mach 3 to Mach 7 flow

5. Data acquisition and processing

ALTITUDE: 80 km

1. **Identification**

organisation	location	designation
Office National d'Etudes et de Recherches Aérospatiales (ONERA)	Châtillon (France)	Laboratoire Aérothermique de Palaiseau A 75/A 611

2. **Contact**

Monsieur le Directeur
29 Avenue de la Division Leclerc
92332 Châtillon Cedex

3. **General Performance**

Type of engine:	Solid or liquid propellants reduced scale motors
Firing orientation:	A 75 Vertical A 611 Horizontal
Test cell size:	A 75: Diam 5 m; Length 15 m A 611: Diam 3 m, Length 20 m
Capacities: mean mass flow:	
max thrust:	
Max firing duration:	depends on flow rate
Transients:	

4. **Technical equipment**

5. **Data acquisition and processing**

ALTITUDE: 25 km

1. Identification

organisation	location	designation
Office National d'Etudes et de Recherches Aérospatiales (ONERA)	Châtillon (France)	Soufflerie de Modane S 4 MA

2. Contact

Monsieur le Directeur
29 Avenue de la Division Leclerc
92332 Châtillon Cedex

3. General Performance

Type of engine:	Solid or liquid propellant motor
Firing orientation:	Horizontal
Test cell size:	Diam. 1.2 m; Length 4.7 m
Capacities: mean mass flow:	
max thrust:	
Max firing duration:	Depends on flow rate
Transients:	

4. Technical equipment

Exhaust: vacuum tank 4000 m^3
Exhaust gas diffuser (subsonic)
Mach 6 flow

5. Data acquisition and processing

ALTITUDE: 25 km

1. Identification

organisation	location	designation
Office National d'Etudes et de Recherches Aérospatiales (ONERA)	Châtillon (France)	Soufflerie de Modane S 4 B

2. Contact

Monsieur le Directeur
29 Avenue de la Division Leclerc
92332 Châtillon Cedex

3. General Performance

Type of engine:	Solid or liquid propellant motor
Firing orientation:	Horizontal
Test cell size:	Diam. 2.5 m; Length 14 m
Capacities: mean mass flow:	
max thrust:	
Max firing duration:	Depends on flow rate
Transients:	

4. Technical equipment

Exhaust: vacuum tank 4000 m^3
Exhaust gas diffuser (subsonic)
Mach 6 flow

5. Data acquisition and processing

ALTITUDE: 25 km

1. Identification

organisation	location	designation
Société Européenne de Propulsion (SEP) Division Propulsion à Liquides — Espace	Vernon (France)	PF 41

2. Contact

Monsieur le Directeur
BP 802
27207 Vernon

3. General Performance

Type of engine:	Cryogenic (LH2/LOX) ARIANE's third stage motor
Firing orientation:	Vertical down
Test cell size:	
Capacities: mean mass flow:	15 kg/s
max thrust:	70 kN
Max firing duration:	900 s
Transients:	Thrust vectoring

4. Technical equipment

Steam ejector for diffuser starting
Exhaust gas diffuser
Nitrogen injector as an anti back-flow device
Lateral turbo-pumps for secondary flow exhaust

5. Data acquisition and processing

ALTITUDE: 35 km

1. Identification

organisation	location	designation
Deutsche Forschungs und Versuchsanstalt für Luft und Raumfahrt (DFVLR)	Hardthausen (Germany)	P 1.5

2. Contact

Professor Lo, Director
D 7101
Hardthausen Am Kocher
Germany

3. General Performance

Type of engine:	Liquid propellant motor
Firing orientation:	Vertical (down/up)
Test cell size:	Diam. 0.8 m; Elevation 1 m
Capacities: mean mass flow:	
max thrust:	0.5 kN
Max firing duration:	4200 s
Transients:	

4. Technical equipment

Steam ejectors (first stage used as an anti back-flow device)
Chemical steam production
Exhaust gas diffusers
Exhaust gas scrubber
Mechanical pumps and oil diffusion ejectors for low, or very low, cell pressures before firing (100 Pa and 0.01 Pa)

5. Data acquisition and processing

See "Test Facility P 4.2"

ALTITUDE: 24 km

1. Identification

organisation	location	designation
Deutsche Forschungs und Versuchsanstalt für Luft und Raumfahrt (DFVLR)	Hardthausen (Germany)	P 3

2. Contact

Professor Lo, Director
D 7101
Hardthausen Am Kocher
Germany

3. General Performance

Type of engine:	Liquid propellant motor
Firing orientation:	Vertical down
Test cell size:	Diam. 2.5 m; Elevation 3.5 m
Capacities: mean mass flow:	
max thrust:	70 kN
Max firing duration:	600 s
Transients:	

4. Technical equipment

Steam ejectors
Chemical steam production
Exhaust gas diffuser

5. Data acquisition and processing

See "Test Facility P 4.2"

ALTITUDE: 24 km

1. Identification

organisation	location	designation
Deutsche Forschungs und Versuchsanstalt für Luft und Raumfahrt (DFVLR)	Hardthausen (Germany)	P 4.2

2. Contact

Professor Lo, Director
D 7101
Hardthausen Am Kocher
Germany

3. General Performance

Type of engine:	Liquid propellant motor (ARIANE's second stage)
Firing orientation:	Vertical down
Test cell size:	Basis 2.6 m × 2.6 m; Elevation 4.6 m
Capacities: mean mass flow:	
max thrust:	800 kN
Max firing duration:	6300 s

Transients:

4. Technical equipment

Steam ejectors (for pre-fire vacuum)
Chemical steam production
Exhaust gas diffuser
Protective shield surrounding the nozzle for heat removal

5. Data acquisition and processing

The total digital acquisition capability is 256 channels at a total sample rate of 40,000 samples/s; each channel sample rate fluctuates from 25 to 500 samples/s. Special high frequency measurements (pressure transients) are recorded on an analog system, with frequencies up to 40 khz. For redundancy in case of computer failure, each measurement is converted into PCM signal and recorded on magnetic tapes. During the test, up to 120 tests readings are indicated to the test personnel at the control console by either analog or digital means.

ALTITUDE: 27 km

1. **Identification**

organisation	location	designation
SNIA-BPD Settore Difesa e Spazio	Collefero (Roma-Italy)	ISA 1

2. **Contact**

Mr Conte
Corso Garibaldi
0034. Collefero (Roma)
Italy

3. **General Performance**

Type of engine:	Solid propellant rocket motor
Firing orientation:	Horizontal
Test cell size:	Diam. 3 m; Length 4 m
Capacities: mean mass flow:	15 kg/s
max thrust:	22 kN
Max firing duration:	60 s

Transients:

4. **Technical equipment**

Steam ejectors (steam stored in accumulators)
Exhaust gas diffuser
Gas cooling by water injection
Isolation gate valve (anti back-flow device)

5. **Data acquisition and processing**

See "Test Facility ISA 2"

ALTITUDE: 27 km

1. **Identification**

organisation	location	designation
SNIA-BPD Settore Difesa e Spazio	Collefero (Roma-Italy)	ISA 2

2. **Contact**

Mr Conte
Corso Garibaldi
0034. Collefero (Roma)
Italy

3. **General Performance**

Type of engine:	Solid propellant rocket motor
Firing orientation:	Horizontal
Test cell size:	Diam. 6 m; Length 10 m
Capacities: mean mass flow:	35 kg/s
max thrust:	50 kN
Max firing duration:	300 s

Transients:

4. **Technical equipment**

Steam ejectors
Chemical steam diffuser
Exhaust gas diffuser
Gas cooling by water injection
Isolation gate valve (anti back-flow device)
Spinning device

5. **Data acquisition and processing**

Monitoring and control system:
* computer G.A. 16/440 and input/output devices
* interactive software (check and actuation of valves; red lines limit check; timing check)
* NEFF analog to digital converter (128 analog inputs)
* digital input/output (96 inputs; 96 outputs)
* process interrupt module (16 digital interrupts)

Data acquisition system:
* computer G.A. 16/440 and input/output devices
* Preston analog to digital converter (64 inputs)
* general purpose 32 inputs/32 outputs
* process interrupt module (16 digital interrupts)

ALTITUDE: 30 km

1. **Identification**

organisation	location	designation
Royal Aircraft Establishment (RAE)	Pyestock (UK)	Cell 2

2. **Contact**

Mr Drabble, Head Trials Section
Pyestock — Farnborough Hants
GU14 0LS — England

3. **General Performance**

Type of engine:	Solid propellant rocket motor
Firing orientation:	Horizontal
Test cell size:	Diam. 4 m; Length 20 m
Capacities: mean mass flow:	10 kg/s
max thrust:	30 kN
Max firing duration:	No limitation
Transients:	

4. **Technical equipment**

Air ejectors (rotating machines)
Exhaust gas diffuser
Gas cooling by water injection
Anti back-flow device: ejectors and isolation gate valves

5. **Data acquisition and processing**

* Steady state: scanning rate 80 samples/s
* Transients: 18 channels on analog continuous recording

6. **Measuring equipment**

* Steady state: 100 pressure transducers
  200 thermocouples
  25 resistance element temperatures
  18 frequencies

* Transient: projected future capability
  40 channels digitally recorded by computer, sampling rate variable from 1 to 200 samples/s. (Later to be improved to 500 sample/s.

ALTITUDE: 50 km

1. **Identification**

organisation	location	designation
Royal Ordnance Explosive Division	Westcott (UK)	6. site Cells 1 and 2

2. **Contact**

Mr Needs
6. Site — ROF Westcott
Westcott — Aylesbury
Bucks — England

3. **General Performance**

Type of engine:	Cell 1 solid propellants Cell 2 liquid propellants
Firing orientation:	Horizontal
Test cell size:	Diam. 4 m; Length 20 m
Capacities: mean mass flow:	0.01 kg/s
max thrust:	49 kN
Max firing duration:	Cell 1: a few seconds Cell 2: no limitation if flow rate is limited to 0.01 kg/s
Transients:	

4. **Technical equipment**

Closed chambers, to be used as separate test cells or linked together to form a single test cell.
Exhaust compressor (rotating)

5. **Data acquisition and processing**

On-line data acquisition (60 kHZ): 16 channels, pressure or thrust.
Analog recording: 12 channels pressure or thrust, 12 channels thermocouples/event marks.
Ignition circuit: low voltage 0—28 volts; high voltage 3 kV

ALTITUDE: 51 km

1. Identification

organisation	location	designation
Air Force Systems Command Arnold Engineering Development Center A.E.D.C.	Arnold Air Force Station Tennessee — USA	Rocket Development Test Cell T-3

2. Contact

Mr Carlton Garner
AECD/DOCA
Arnold AFS, TN 37389-9998

3. General Performance

Type of engine:	Solid propellant rocket motor
Firing orientation:	Horizontal
Test cell size:	Diam. 3.1 m; Length 4.9 m
Capacities: mean mass flow:	30 kg/s
max thrust:	89 kN
Max firing duration:	100 s
Transients:	Spin rig

4. Technical equipment

Exhaust gas diffuser
Exhaust gas scrubber
Temperature conditioning (−12 C; +43 C)
2-Stage steam ejector
3-Stage exhaust compressors (rotating)

5. Data acquisition and processing

Computer controlled data acquisition system: 50,000 measurements/s; includes alpha-numeric displays, analog recorders, movie cameras and television systems.

6. Measuring equipment

Thrust		91 kN
Digital acquisition	(141 total)	4 strain gauge transducers 20 strain gauges 44 thermocouples 20 general purpose transducers 1 timing
Analog acquisition	36 FM MUX 18 O'graph 54 total	Piezo electric transd. Thrust vector control Ignition lines 2 Video

7. Accuracy

measurement	%	range
thrust. kN (lbf)	0.3	91 — 182 (10 000 — 20 000)
propellant expanded mass Kg (lbm)	0.1	1 020 — 2 045 (2 250 — 4 500)
combustion pressure MPa (psia)	0.7	3.4 — 6.8 (500 — 1000)
vacuum pressure KPa (psia)	.006	.07 — 0.7 (0.1 — 1.0)
I_{sv} .sec	0.3	

ALTITUDE: 30 km

1. Identification

organisation	location	designation
Air Force Systems Command Arnold Engineering Development Center A.E.D.C.	Arnold Air Force Station Tennessee — USA	Rocket Development Test Cell J-3

2. Contact

Mr Gene Sanders
A.E.D.C./D.O.
Arnold AFS, TN 37389-9998

3. General Performance

Type of engine:	Liquid/Solid propellant rocket motors
Firing orientation:	Vertical down
Test cell size:	Diam. 5.1 m; Length 12 m
Capacities: mean mass flow:	145 kg/s
max thrust:	500 kN
Max firing duration:	40 s (reduced to 40 s by propellants quantity limitation)
Transients:	Exit cone extension Thrust vectoring

4. Technical equipment

Temperature conditioning (0°C; +45°C)
Exhaust gas diffuser
Steam ejector
Exhaust gas scrubber
4-stage exhaust compressors (rotating)
Motor quench

5. Data acquisition and processing

Computer controlled data acquisition system: 80,000 measurements/s; includes x-y recorders, movie cameras and television systems.

6. Measuring equipment

Thrust		500 kN
Digital acquisition	200 total	500 kN strain gauge transducers strain gauge thermocouples general purpose transducers 96 timing
Analog acquisition	144 FM MUX 64 O'graph 208 total	piezo electric transducers thrust vector control ignition lines

7. Accuracy

measurement	%	range
thrust. kN (lbf)	0.75 −1.0	0 — 500 (0 — 100,000)
propellant expanded mass Kg (lbm)	1.0	0 — 73 (0 — 160)
combustion pressure MPa (psia)	0.7	5.1 — 13.6 (750 — 2000)
vacuum pressure KPa (psia)	0.006 psi	.07 — 0.7 (0.1 — 1.0)
I_{sv} .sec	1.0	

ALTITUDE: 30 km

1. Identification

organisation	location	designation
Air Force Systems Command Arnold Engineering Development Center A.E.D.C.	Arnold Air Force Station Tennessee — USA	Rocket Development Test Cell J-4

2. Contact

Mr Gene Sanders
A.E.D.C./D.O.
Arnold AFS, TN 37389-9998

3. General Performance

Type of engine:	Liquid/Solid propellant rocket motors
Firing orientation:	Vertical down
Test cell size:	Diam. 15 m; Length 22 m
Capacities: mean mass flow:	455 kg/s
max thrust:	2300 kN
Max firing duration:	60 s
Transients:	Exit cone extension Thrust vectoring

4. Technical equipment

Basic design for 7000 kN (thrust) with modifications
Exhaust gas diffuser
Annular steam ejector
Temperature conditioning (2°C—40°C)
Parallel bypass 3-stage steam ejector
Exhaust gas scrubber/accumulator
4-stage exhaust compressors (rotating)

5. Data acquisition and processing

Computer controlled data acquisition system (DACS): 250,000 measurements/s; includes real-time graphics display, altitude control system (ACS), video display, movie cameras, and television recording systems.

6. Measuring equipment

Thrust		2300 kN
Digital acquisition	*246 combined	*strain gauge transducers *strain gauge 96 thermocouples *general purpose transducers 96 timing
Analog acquisition	74 FM MUX 24 O'graph 98 total	piezo electric transducers thrust vector control ignition lines 15 video

7. Accuracy

measurement	%	range
thrust. kN (lbf)	0.75	0 — 2300 (0 — 500,000)
propellant expanded mass Kg (lbm)	1.0	0 — 455 (0 — 1000)
combustion pressure MPa (psia)	0.7	5.1 — 13.6 (750 — 2000)
vacuum pressure KPa (psia)	0.006 psi	.07 — 0.7 (0.1 — 1.0)
I_{sv} .sec	1.0	

ALTITUDE: 45 km

1. Identification

organisation	location	designation
Air Force Systems Command Arnold Engineering Development Center A.E.D.C.	Arnold Air Force Station Tennessee — USA	Rocket Development Test Cell J-5

2. Contact

Mr Gene Sanders
A.E.D.C./D.O.
Arnold AFS, TN 37389-9998

3. General Performance

Type of engine:	Solid propellant rocket motor
Firing orientation:	Horizontal
Test cell size:	Diam. 5 m; Length 15 m
Capacities: mean mass flow:	195 kg/s
max thrust:	570 kN
Max firing duration:	70 s
Transients:	Exit cone extension Thrust vectoring Spin rig

4. Technical equipment

Temperature conditioning (2C—45C)
Exhaust gas diffuser
Annular steam ejector
Exhaust gas scrubber
Parallel bypass 2-stage steam ejector
Motor quench
4-stage exhaust compressors (rotating)

5. Data acquisition and processing

Computer controlled data acquisition system (DACS): 250,000 measurements/s; includes real-time graphics displays, movie cameras and television recording systems.

6. Measuring equipment

Thrust		570 kN
Digital acquisition	184 combined	strain gauge transducers strain gauge 64 thermocouples general purpose transducers 128 timing
Analog acquisition	72 FM MUX 24 O'graph 96 total	piezo electric transducers thrust vector control ignition lines 13 video

7. Accuracy

measurement	%	range
thrust. kN (lbf)	0.20—0.5	0 — 570 (0—125,000)
propellant expanded mass Kg (lbm)	0.1	7 000 — 11,500 (15 000 — 25,000)
combustion pressure MPa (psia)	0.7	5.1 — 10.2 (750 — 1500)
vacuum pressure KPa (psia)	.006 psi	.07 — 0.7 (0.1 — 1.0)
I_{sv} .sec	0.2	

ALTITUDE: 33 km

1. Identification

organisation	location	designation
Air Force Rocket Propulsion Laboratory (A.F.R.P.L.)	Edwards Air Force Base California — USA	Area 1—42 Cell A

2. Contact

Mr Hart, Manager
AFRPL/Edwards AFB
CA 93 535 — USA

3. General Performance

Type of engine:	Solid propellant rocket motor
Firing orientation:	Horizontal
Test cell size:	Diam. 3.7 m; Length 8.5 m
Capacities: mean mass flow:	75 kg/s
max thrust:	220 kN
Max firing duration:	300 s
Transients:	Exit cone extension Thrust vectoring

4. Technical equipment

Temperature conditioning
Spinning device
Exhaust gas diffuser
Steam ejectors (3 parallel sets of 2 serial ejectors)
Both chemical steam generation and stored boiled water in accumulators
Exhaust gas scrubber

5. Data acquisition and processing

One control room for the 3 cells A, B, D (area 1.42), located at about 100 m from the nearest (cell D).

6. Measuring equipment

Thrust	2 channels
Digital acquisition	96 strain gages (or pressure) 96 voltage channels (flow rate or temperature) 18 timing channels Total: 50 000 samples/s.
Analog acquisition	24 FM channels (8 kHz per channel) 2 ignition systems 8 video channels (with pan and tilt mechanisms)
Output devices	One 18 tracks oscillograph
Movie cameras	Up to 11 000 frames/s.

7. Accuracy

measurement	%	range
thrust. kN (lbf)	0.1	Up to 220 (50 000)
propellant expanded mass Kg (lbm)	0.1	30 — 7 000 (66 — 15 500)
combustion pressure MPa (psia)	0.5	0 — 14 (0 — 2000)
vacuum pressure KPa (psia)	1.0	0.7 — 2 (0.1 — 0.3)
I_{sv} .sec	0.25— 0.5	#300 s

ALTITUDE: 33 km

1. Identification

organisation	location	designation
Air Force Rocket Propulsion Laboratory (A.F.R.P.L.)	Edwards Air Force Base California — USA	Area 1—42 Cell B

2. Contact

Mr Hart, Manager
AFRPL/Edwards AFB
CA 93 535 — USA

3. General Performance

Type of engine:	Liquid propellant rocket motor
Firing orientation:	Vertical
Test cell size:	Diam. 4.9 m; Elevation 8.5 m
Capacities: mean mass flow:	65 kg/s
max thrust:	220 kN
Max firing duration:	300 s
Transients:	Exit cone extension

4. Technical equipment

Temperature conditioning
Exhaust gas diffuser
Steam ejectors (3 parallel sets of 2 serial ejectors)
Both chemical steam generation and stored boiled water in accumulators
Exhaust gas scrubber
Isolation valve

5. Data acquisition and processing

One control room for the 3 cells A, B, D (area 1.42), located at about 100 m from the nearest (cell D).

6. Measuring equipment

Thrust		2 channels
Digital acquisition		96 strain gauges (or pressure)
		96 voltage channels (flow rate or temperature)
		18 timing channels
	Total:	50 000 samples/s.
Analog acquisition		24 FM channels (8 kHz per channel)
		8 video channels (with pan and tilt mechanisms)
Output devices		One 18 tracks oscillograph
Movie cameras		Up to 11 000 frames/s.

7. Accuracy

measurement	%	range
thrust. kN (lbf)	0.1	Up to 220 (50 000)
propellant expanded mass Kg (lbm)	0.1	30 — 7 000 (66 — 15 500)
combustion pressure MPa (psia)	0.5	0 — 14 (0 — 2000)
vacuum pressure KPa (psia)	1.0	0.7 — 2 (0.1 — 0.3)
I_{sv} .sec	0.25— 0.5	#300 s

ALTITUDE: 33 km

1. Identification

organisation	location	designation
Air Force Rocket Propulsion Laboratory (A.F.R.P.L.)	Edwards Air Force Base California — USA	Area 1—42 Cell D

2. Contact

Mr Hart, Manager
AFRPL/Edwards AFB
CA 93 535 — USA

3. General Performance

Type of engine:	Solid propellant rocket motor
Firing orientation:	Horizontal
Test cell size:	Diam. 3.2 m; Elevation 7.9 m
Capacities: mean mass flow:	30 kg/s
max thrust:	90 kN
Max firing duration:	300 s
Transients:	Exit cone extension Thrust vectoring

4. Technical equipment

Temperature conditioning
Spinning device
Exhaust gas diffuser
Steam ejectors (3 parallel sets of 2 serial ejectors)
Both chemical steam generation and stored boiled water in accumulators
Exhaust gas scrubber
Isolation valve

5. Data acquisition and processing

One control room for the 3 cells A, B, D (area 1.42), located at about 100 m from the nearest (cell D).

6. Measuring equipment

Thrust	2 channels
Digital acquisition	96 strain gauges (or pressure) 96 voltage channels (flow rate or temperature) 18 timing channels Total: 50 000 samples/s.
Analog acquisition	24 FM channels (8 kHz per channel) 8 video channels (with pan and tilt mechanisms)
Output devices	One 18 tracks oscillograph
Movie cameras	Up to 11 000 frames/s.

7. Accuracy

measurement	%	range
thrust. kN (lbf)	0.1	Up to 220 (50 000)
propellant expanded mass Kg (lbm)	0.1	30 — 7 000 (66 — 15 500)
combustion pressure MPa (psia)	0.5	0 — 14 (0 — 2000)
vacuum pressure KPa (psia)	1.0	0.7 — 2 (0.1 — 0.3)
I_{sv} .sec	0.25— 0.5	≠300 s

ALTITUDE: 30 km

1. Identification

organisation	location	designation
Air Force Rocket Propulsion Laboratory (A.F.R.P.L.)	Edwards Air Force Base California — USA	Area 1—14 Cell A

2. Contact

Mr Hart, Manager
AFRPL/Edwards AFB
CA 93 535 — USA

3. General Performance

Type of engine:	Liquid propellant rocket motor
Firing orientation:	Horizontal
Test cell size:	Diam. 2.6 m; Length 9.3 m
Capacities: mean mass flow:	
max thrust:	22 kN (abutment); 4 kN (thrust stand)
Max firing duration:	Up to 1800 s, dependent on thrust
Transients:	

4. Technical equipment

Temperature conditioning
Thrust stand independent from cell chamber
Exhaust gas diffuser
Exhaust gas scrubber
Mechanical pumps when required
Steam ejectors (chemical steam generation)

5. Data acquisition and processing

Two control rooms for the 4 cells A, C, D, E (area 1.14), so that several facilities can run at the same time.

Each system is composed of:

— Digital: 192 channels at a maximum 64000 samples/s rate.
Any channel can sample up to 2000 samples/s.
A choice must be made between:
- 11 flow channels
- 132 temperature channels
- 64 strain gauges

— Analog: 1 FM system of 14 channels
Frequency response: 14 kHz
Accuracy: 1% of full scale.

— Movie cameras and TV when required.

ALTITUDE: 40 km

1. **Identification**

organisation	location	designation
Air Force Rocket Propulsion Laboratory (A.F.R.P.L.)	Edwards Air Force Base California — USA	Area 1—14 Cell C

2. **Contact**

Mr Hart, Manager
AFRPL/Edwards AFB
CA 93 535 — USA

3. **General Performance**

Type of engine:	Liquid propellant rocket motor
Firing orientation:	Horizontal
Test cell size:	Diam. 2.1 m; Length 6 m
Capacities: mean mass flow:	
max thrust:	0.4 kN (abutment) + vertical abutment
Max firing duration:	Continuous (mechanical pumps)
Transients:	

4. **Technical equipment**

Temperature conditioning
Thrust stand independent from cell chamber
Exhaust gas diffuser
Exhaust gas scrubber
Mechanical pumps
Connection to oil-diffusion pumps when required (very low cell ressure levels)

5. **Data acquisition and processing**

Two control rooms for the 4 cells A, C, D, E (area 1.14), so that several facilities can run at the same time.

Each system is composed of:

— Digital: 192 channels at a maximum 64000 samples/s rate.
Any channel can sample up to 2000 samples/s.
A choice must be made between:
- 11 flow channels
- 132 temperature channels
- 64 strain gauges

— Analog: 1 FM system of 14 channels
Frequency response: 14 kHz
Accuracy: 1% of full scale.

— Movie cameras and TV when required.

ALTITUDE: 40 km

1. Identification

organisation	location	designation
Air Force Rocket Propulsion Laboratory (A.F.R.P.L.)	Edwards Air Force Base California — USA	Area 1—14 Cell D

2. Contact

Mr Hart, Manager
AFRPL/Edwards AFB
CA 93 535 — USA

3. General Performance

Type of engine:	Liquid propellant rocket motor
Firing orientation:	Horizontal
Test cell size:	Diam. 2.4 m; Length 4.8 m
Capacities: mean mass flow:	
max thrust:	6.6 kN (abutment) 1.2 kN (thrust stand)
Max firing duration:	— Continuous (mechanical pumps) if thrust <40 N — Up to 900 s (steam ejectors)
Transients:	

4. Technical equipment

Temperature conditioning
Thrust stand independent from cell chamber
Exhaust gas diffuser
Exhaust gas scrubber
Steam ejectors (chemical steam generation)
Mechanical pumps when required

5. Data acquisition and processing

Two control rooms for the 4 cells A, C, D, E (area 1.14), so that several facilities can run at the same time.

Each system is composed of:

— Digital: 192 channels at a maximum 64000 samples/s rate.
Any channel can sample up to 2000 samples/s.
A choice must be made between:
- 11 flow channels
- 132 temperature channels
- 64 strain gauges

— Analog: 1 FM system of 14 channels
Frequency response: 14 kHz
Accuracy: 1% of full scale.

ALTITUDE: 40 km

1. Identification

organisation	location	designation
Air Force Rocket Propulsion Laboratory (A.F.R.P.L.)	Edwards Air Force Base California — USA	Area 1—14 Cell E

2. Contact

Mr Hart, Manager
AFRPL/Edwards AFB
CA 93 535 — USA

3. General Performance

Type of engine:	Liquid propellant rocket motor
Firing orientation:	Vertical down
Test cell size:	Diam. 2.7 m; Elevation 6 m
Capacities: mean mass flow:	
max thrust:	22 kN (abutment) 1.2 kN (thrust stand)
Max firing duration:	— Continuous (mechanical pumps) if thrust <40 N — Up to 900 s (steam ejectors)
Transients:	

4. Technical equipment

Temperature conditioning
Thrust stand independent from cell chamber
Exhaust gas diffuser
Exhaust gas scrubber
Steam ejectors (chemical steam generation)
Mechanical pumps when required

5. Data acquisition and processing

Two control rooms for the 4 cells A, C, D, E (area 1.14), so that several facilities can run at the same time.

Each system is composed of:

— Digital: 192 channels at a maximum 64000 samples/s rate.
Any channel can sample up to 2000 samples/s.
A choice must be made between:
- 11 flow channels
- 132 temperature channels
- 64 strain gauges

— Analog: 1 FM system of 14 channels
Frequency response: 14 kHz
Accuracy: 1% of full scale.

ALTITUDE: 57 km

1. Identification

organisation	location	designation
Aerojet Techsystems Company	Sacramento, California — USA	Test cell J 3

2. Contact

Mr Brincka, Director, Test Operations
P.O. Box 13 222 — SACRAMENTO
CALIFORNIA 95 813 — USA

3. General Performance

Type of engine:	Solid, liquid, cryogenic propellants
Firing orientation:	Horizontal
Test cell size:	Diam. 2.4 m; Length 39 m
Capacities: mean mass flow:	
max thrust:	7 kN
Max firing duration:	3400 s with 3-stages ejectors: (30 km without diffuser, 57 with one) 16 Hours with 2-stages ejectors: (24 km without diffuser, 30 with one)
Transients:	pulse mode operation thermal soak-back mission duty cycle

4. Technical equipment

Temperature conditioning
Exhaust gas diffuser when required
Isolation valve (anti back-flow device)
Exhaust gas scrubber
Steam ejectors (accumulators)

5. Data acquisition and processing

The control room is located adjacent to the “J” area office. Provision is made for pre-test calibration; data acquisition, display, and recordation; interfacing with the Central Data Recording Facility; and data reduction and analysis. Pressure, temperature, flow and speed, and high-frequency parameters are displayed visually, as well as being recorded on strip charts, Visicorders, and analog and digital tape. “On-line” performance data are available while a test is in progress. The sampling rate for digital data is approximately 40 or 20 samples/s per channel. Time between samples is programmable up to a maximum of 20 000 samples/s.

ALTITUDE: 30 km

1. Identification

organisation	location	designation
Aerojet Techsystems Company	Sacramento, California — USA	Test cell J 4

2. Contact

Mr Brincka, Director, Test Operations
P.O. Box 13 222 — SACRAMENTO
CALIFORNIA 95 813 — USA

3. General Performance

Type of engine:	Solid, liquid, cryogenic propellants
Firing orientation:	Horizontal
Test cell size:	4.5 m * 5.3 m * 3 m
Capacities: mean mass flow:	
max thrust:	90 kN
Max firing duration:	Up to 3400 s; depends on thrust and simulated altitude: a 25 kN thrust system can be tested at 45 km during 50 s; currently, a 30 kN thrust system can be tested at 30 km during 200 s.
Transients:	Exit cone extension thrust vectoring pulse mode operation thermal soak-back mission duty cycle

4. Technical equipment

Temperature conditioning
Spinning device
Exhaust gas diffuser
Isolation valve (anti back-flow device)
Exhaust gas scrubber
Steam ejectors (accumulators)

5. Data acquisition and processing

The control room is located adjacent to the "J" area office. Provision is made for pre-test calibration; data acquisition, display, and recordation; interfacing with the Central Data Recording Facility; and data reduction and analysis. Pressure, temperature, flow and speed, and high-frequency parameters are displayed visually, as well as being recorded on strip charts, Visicorders, and analog and digital tape. "On-line" performance data are available while a test is in progress. The sampling rate for digital data is approximately 40 or 20 samples/s per channel. Time between samples is programmable up to a maximum of 20 000 samples/s.

ALTITUDE: 40 km

1. Identification

organisation	location	designation
The Marquardt Company Marquardt Jet Laboratory (M.J.L.)	Van Nuys California — USA	Cell 1

2. Contact

Mr Sund, Director Engineering
16 555 Saticoy Street — VAN NUYS
CALIFORNIA 91 409 — USA

3. General Performance

Type of engine:	Liquid propellant rocket motor
Firing orientation:	Horizontal
Test cell size:	Diam 1.2 m; Length 1.1 m
Capacities: mean mass flow:	
max thrust:	5.5 kN
Max firing duration:	limited by propellant supply
Transients:	Thrust vectoring

4. Technical equipment

Isolation valve
Exhaust gas diffuser
Exhaust gas scrubber
Steam ejectors (accumulators)

5. Data acquisition and processing

The digital data acquisition systems consists of 36 channels of Neff signal conditioning computer controlled and recorded on magnetic tape. Analog data is recorded on strip chart recorders and O'graph. These instruments along with gauges are also used as quick look data to monitor and control the cell operation. High frequency data is recorded on a FM magnetic recorder, and can be plotted on a x-y plotter.

Data parameters consist of: Propellant valves voltage and current; Engine thrust; Engine chamber pressure; Propellant manifold pressures; Engine and propellant temperatures; Propellant flowrate.

The digital data acquired on magnetic tape is reduced to engineering units and specialized performance calculations made after each run to determine the rocket engine performance. The data stored on magnetic tape, strip charts and O'graph records is retained until satisfaction of the contract, thus making the engine performance data available at any time.

ALTITUDE: 40 km

1. Identification

organisation	location	designation
The Marquardt Company Marquardt Jet Laboratory (M.J.L.)	Van Nuys California — USA	Cell 2 Cell 8

2. Contact

Mr Sund, Director Engineering
16 555 Saticoy Street — VAN NUYS
CALIFORNIA 91 409 — USA

3. General Performance

Type of engine:	Liquid propellant rocket motor
Firing orientation:	Vertical or horizontal
Test cell size:	Diam 4 m; Length 6 m
Capacities: mean mass flow:	
max thrust:	180 kN
Max firing duration:	limited by propellant supply
Transients:	Thrust vectoring

4. Technical equipment

Exhaust gas diffuser
Exhaust gas scrubber
Steam ejectors (accumulators)
Cells 2 and 8 are ramjet test cells (see AGARDograph 269)

5. Data acquisition and processing

Analog channels: 240 Max freq: 40 kHz (*1)
Digital channels: 240 Max sample rate: 10 kHz (*2)
Real-time display channels: 8. Real-time data reduction: 253 (*3)
Electrical controls: 16 computer-generated control functions

(*1) Analog signals:

a. The maximum frequency response of 40 kHz is attainable when piezo-electric transducers and pickups are used with FM magnetic recording.

b. Medium frequency response, up to 5 kHz, is obtainable when straingauge type transducers and light-beam type recorders are used.

c. Low frequency response 1—2 kHz is obtained when strip-charts are used for data recording.

(*2)
The in-cell digital data acquisition system has 240 channels and set at 10 kHz sampling rate. Channels are randomly selected in sequence determined by test and frequency response requirements.

(*3)
All test parameters, up to 256, are reduced to engineering units in real time by the data acquisition computer. Eight channels, as selected by test requirements, are displayed in real time in the control room.

ALTITUDE: 71 km

1. Identification

organisation	location	designation
The Marquardt Company Marquardt Jet Laboratory (M.J.L.)	Van Nuys California — USA	Cell 9 #1

2. Contact

Mr Sund, Director Engineering
16 555 Saticoy Street — VAN NUYS
CALIFORNIA 91 409 — USA

3. General Performance

Type of engine:	Liquid propellant rocket motor
Firing orientation:	Vertical down
Test cell size:	6 m diameter sphere
Capacities: mean mass flow:	
max thrust:	0.05 kN
Max firing duration:	No limitation
Transients:	Thermal space soak

4. Technical equipment

Isolation valve
Steam ejectors (accumulators)
Space soak achieved by use of cryopanels

5. Data acquisition and processing

The digital data acquisition systems consists of 36 channels of Neff signal conditioning computer controlled and recorded on magnetic tape. Analog data is recorded on strip chart recorders and O'graph. These instruments along with gauges are also used as quick look data to monitor and control the cells operation. High frequency data is recorded on a FM magnetic recorder, and can be plotted on a x-y plotter.

Data parameters consist of: Propellant valves voltage and current; Engine thrust; Engine chamber pressure; Propellant manifold pressures; Engine and propellant temperatures; Propellant flowrate.

The digital data acquired on magnetic tape is reduced to engineering units and specialized performance calculations made after each run to determine the rocket engine performance. The data stored on magnetic tape, strip charts and O'graph records is retained until satisfaction of the contract, thus making the engine performance data available at any time.

ALTITUDE: 71 km

1. Identification

organisation	location	designation
The Marquardt Company Marquardt Jet Laboratory (M.J.L.)	Van Nuys California — USA	Cell 9 # 2

2. Contact

Mr Sund, Director Engineering
16 555 Saticoy Street — VAN NUYS
CALIFORNIA 91 409 — USA

3. General Performance

Type of engine:	Liquid propellant rocket motor
Firing orientation:	Horizontal
Test cell size:	6 m diameter sphere
Capacities: mean mass flow:	
max thrust:	5 kN
Max firing duration:	No limitation
Transients:	Thermal space soak

4. Technical equipment

Isolation valve
Steam ejectors (accumulators)
Space soak achieved by use of cryopanels
Exhaust gas scrubber

5. Data acquisition and processing

The digital data acquisition systems consists of 36 channels of Neff signal conditioning computer controlled and recorded on magnetic tape. Analog data is recorded on strip chart recorders and O'graph. These instruments along with gauges are also used as quick look data to monitor and control the cells operation. High frequency data is recorded on a FM magnetic recorder, and can be plotted on a x-y plotter.

Data parameters consist of: Propellant valves voltage and current; Engine thrust; Engine chamber pressure; Propellant manifold pressures; Engine and propellant temperatures; Propellant flowrate.

The digital data acquired on magnetic tape is reduced to engineering units and specialized performance calculations made after each run to determine the rocket engine performance. The data stored on magnetic tape, strip charts and O'graph records is retained until satisfaction of the contract, thus making the engine performance data available at any time.

ALTITUDE: 40 km

1. Identification

organisation	location	designation
The Marquardt Company Marquardt Jet Laboratory (M.J.L.)	Van Nuys California — USA	Precision rocket laboratory (PRL) Pads D and E

2. Contact

Mr Sund, Director Engineering
16 555 Saticoy Street — VAN NUYS
CALIFORNIA 91 409 — USA

3. General Performance

Type of engine:	Liquid propellant rocket motor
Firing orientation:	Vertical down
Test cell size:	Diam. 1.2 m; Elevation 1 m
Capacities: mean mass flow:	
max thrust:	0.5 kN
Max firing duration:	Limited by propellant supply
Transients:	Thermal space soak

4. Technical equipment

Steam ejectors (accumulators)
Space soak achieved by use of cryopanels
Exhaust gas scrubber

5. Data acquisition and processing

Instrumentation for determination of engine performance consists of *analog* recording only.

The analog data with it associated B & F signal conditioning equipment is recorded on strip chart recorders and O'graph. These instruments along with gauges are used as quick look data to monitor and control the cell operation.

Data parameters consist of: Propellant valves voltage and current; Engine thrust; Engine chamber pressure; Propellant manifold pressures; Engine and propellant temperatures; Propellant flowrate.

The analog data acquired on recorder charts is used to perform calculations to determine the rocket engine performance and these records are retained until satisfaction of the contract, thus making the engine performance data readily available.

ALTITUDE: 75 km

1. Identification

organisation	location	designation
National Aeronautics and Space Administration (NASA) Johnson Space Center White Sands Test Facility (W.S.T.F.)	Las Cruces (New Mexico) USA	TS 302

2. Contact

Mr Rob R.Tillett, Manager
White Sands Test Facility
P.O. Drawer M M
Las Cruces, New Mexico 88004

3. General Performance

Type of engine:	Solid/liquid propellants motors
Firing orientation:	
Test cell size:	Diam. 10.5 m; Elevation 16 m
Capacities: mean mass flow:	
max thrust:	
Max firing duration:	Depends on engine size and installed pumping capability
Transients:	

4. Technical equipment

Temperature conditioning (5C — 50C)
Exhaust gas diffuser can be installed at the bottom of test cell
Exhaust compressor (rotating)

5. Data acquisition and processing

— The system throughput rate is 80 kHz raw data words (16 bits/word) with programmable samples rates.
— 126 analogue channels at 2 kHz.
— 575 digital channels at 1 kHz, multiplexable.
— 145 real time display channels.
— Colour graphics interactive terminals provide real time data and control functions displayed on system fluid schematics.
Near real time quick-look reports are provided by a line printer and high speed plotter.
— Photography: close-circuit TV systems; movie cameras.

ALTITUDE: 35 km

1. Identification

organisation	location	designation
National Aeronautics and Space Administration (NASA) Johnson Space Center White Sands Test Facility (W.S.T.F.)	Las Cruces (New Mexico) USA	TS 401

2. Contact

Mr Rob R.Tillett, Manager
White Sands Test Facility
P.O. Drawer M M
Las Cruces, New Mexico 88004

3. General Performance

Type of engine:	Liquid/cryogenic propellant motors
Firing orientation:	Vertical
Test cell size:	Diam. 10.5 m; Elevation 10 m
Capacities: mean mass flow:	
max thrust:	100 kN (Liquid) 60 kN (cryogenic)
Max firing duration:	Depends on steam flow to ejectors: 80 kg/s = 145 minutes 160 kg/s = 70 minutes 250 kg/s = 45 minutes
Transients:	

4. Technical equipment

Temperature conditioning (5C — 45C)
Centre body exhaust gas diffuser
Steam ejectors (chemical steam production)
Isolation valve

5. Data acquisition and processing

— The system throughput rate is 80 kHz raw data words (16 bits/word) with programmable samples rates.
— 260 analogue channels at 2 kHz.
— 575 digital channels at 1 kHz, multiplexable.
— 145 real time display channels.
— Colour graphics interactive terminals provide real time data and control functions displayed on system fluid schematics.
Near real time quick-look reports are provided by a line printer and high speed plotter.
— Photography: close-circuit TV systems; movie cameras.

ALTITUDE: 35 km

1. Identification

organisation	location	designation
National Aeronautics and Space Administration (NASA) Johnson Space Center White Sands Test Facility (W.S.T.F.)	Las Cruces (New Mexico) USA	TS 403

2. Contact

Mr Rob R.Tillett, Manager
White Sands Test Facility
P.O. Drawer M M
Las Cruces, New Mexico 88004

3. General Performance

Type of engine:	Liquid propellant motors
Firing orientation:	Vertical
Test cell size:	Diam. 10.5 m; Elevation 10 m
Capacities: mean mass flow:	
max thrust:	100 kN
Max firing duration:	Depends on steam flow to ejectors: 80 kg/s = 145 minutes; 160 kg/s = 70 minutes; 250 kg/s = 45 minutes
Transients:	

4. Technical equipment

Temperature conditioning (5C — 45C)
Centre body exhaust gas diffuser
Steam ejectors (chemical steam production)
Isolation valve

5. Data acquisition and processing

— The system throughput rate is 80 kHz raw data words (16 bits/word) with programmable samples rates.
— 260 analogue channels at 2 kHz.
— 575 digital channels at 1 kHz, multiplexable.
— 145 real time display channels.
— Colour graphics interactive terminals provide real time data and control functions displayed on system fluid schematics.
Near real time quick-look reports are provided by a line printer and high speed plotter.
— Photography: close-circuit TV systems; movie cameras.

ALTITUDE: 30 km

1. Identification

organisation	location	designation
National Aeronautics and Space Administration (NASA) Johnson Space Center White Sands Test Facility (W.S.T.F.)	Las Cruces (New Mexico) USA	TS 405

2. Contact

Mr Rob R.Tillett, Manager
White Sands Test Facility
P.O. Drawer M M
Las Cruces, New Mexico 88004

3. General Performance

Type of engine:	Solid propellant motors
Firing orientation:	Horizontal
Test cell size:	Diam. 3 m; Length 7.5 m
Capacities: mean mass flow:	
max thrust:	90 kN
Max firing duration:	Depends on steam flow to ejectors: 80 kg/s = 145 minutes 160 kg/s = 70 minutes 250 kg/s = 45 minutes
Transients:	

4. Technical equipment

Exhaust gas diffuser
Steam ejectors (chemical steam production)
Isolation valve
Spinning device

5. Data acquisition and processing

— The system throughput rate is 80 kHz raw data words (16 bits/word) with programmable samples rates.
— 260 analogue channels at 2 kHz.
— 575 digital channels at 1 kHz, multiplexable.
— 145 real time display channels.
— Colour graphics interactive terminals provide real time data and control functions displayed on system fluid schematics.
Near real time quick-look reports are provided by a line printer and high speed plotter.
— Photography: close-circuit TV systems; movie cameras.

4. APPENDICES

4.1 ABBREVIATIONS

A* = Nozzle Throat Area
Ad = Duct (ejector-diffuser) Entrance Area
Ane = Nozzle Exit Area
Asd = Subsonic Diffuser Exit Area
Ast = Second Throat (ejector-diffuser) Area

Cf = Thrust Coefficient

D* = Nozzle Throat Diameter
Dd = Duct (ejector-diffuser) Entrance Diameter
Dne = Nozzle Exit Diameter
Dsd = Subsonic Diffuser Exit Diameter
Dst = Second Throat (ejector-diffuser) Diameter

IFV = Vacuum Total Impulse
ISV = Vacuum Specific Impulse

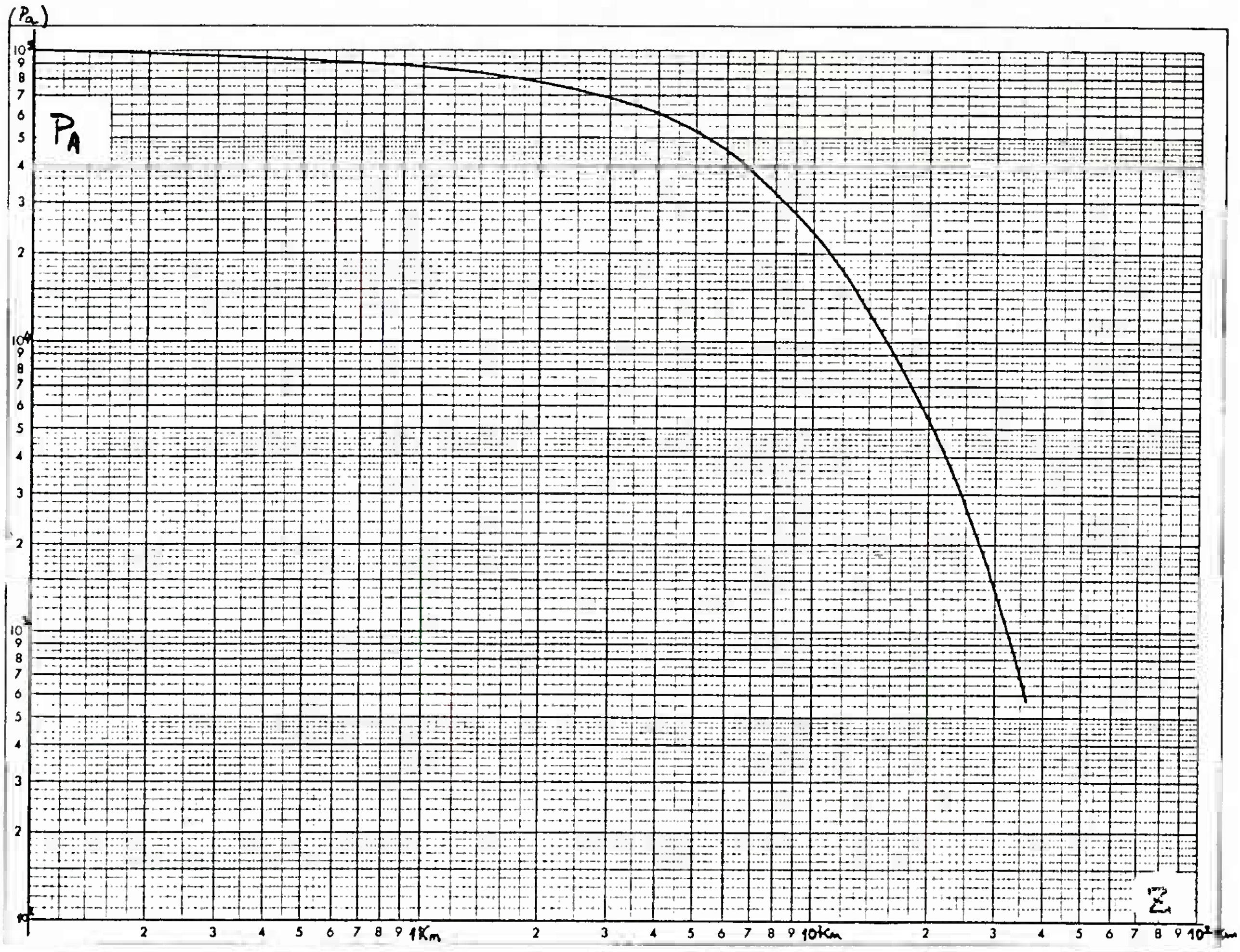
L = Nozzle Exit to Ramp Section Area Distance
Lst = Second Throat (ejector-diffuser) Length

Mf = Mass (of vehicle) After Firing
Mi = Mass (of vehicle) Before Firing

Pa = Ambient Pressure
Pc = Combustion Pressure
Pde = Supersonic Diffuser Exit Static Pressure
Pne = Nozzle Exit Static Pressure
Pt = Normal Shock Static Pressure
(Pt1 = upstream; Pt2 = downstream)

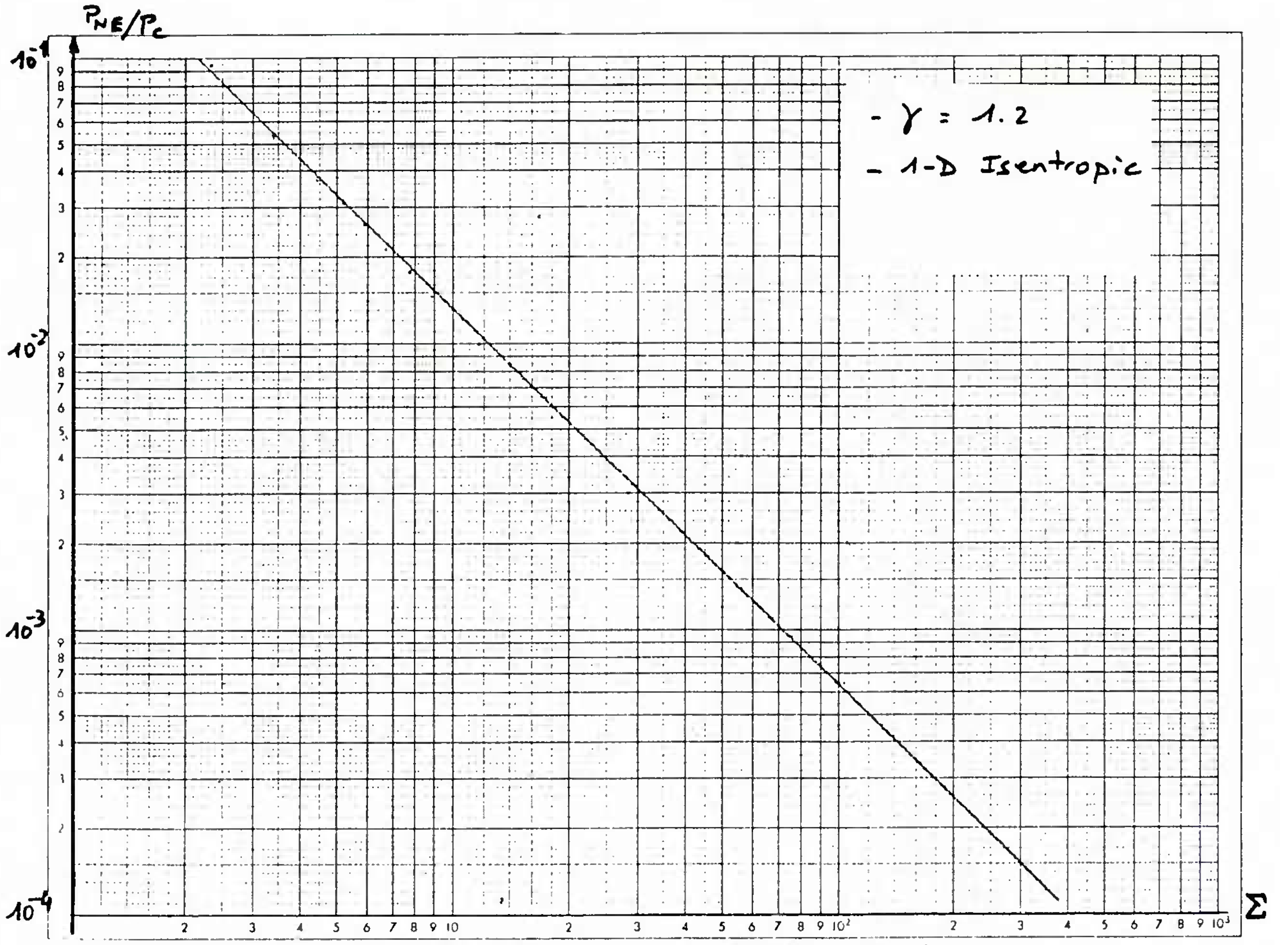
V = Velocity

Δ = Difference (ex: ΔV = Vehicle Velocity Increase)
γ = Ratio of Specific Heats
Λ = Ad/A* = Total Expansion Ratio
ρ = Impingement Angle
ψ = Ast/Ad = Contraction Ratio of a Second Throat Diffuser
θr = Ramp Section Angle
θst = Subsonic Diffuser Angle
Σ = Ane/A* = Nozzle Expansion Ratio
σ = Ad/Ane = Free Jet Expansion Ratio



Pressure vs Altitude
(Pa) (km)

Fig.1



P_{NE}/P_c versus $\Sigma\,(A_{NE}/A_*)$

Fig.2

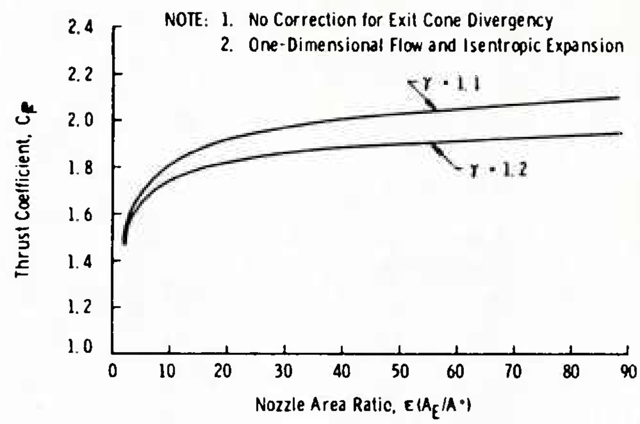


Fig.3 Thrust coefficient variation with nozzle area ratio

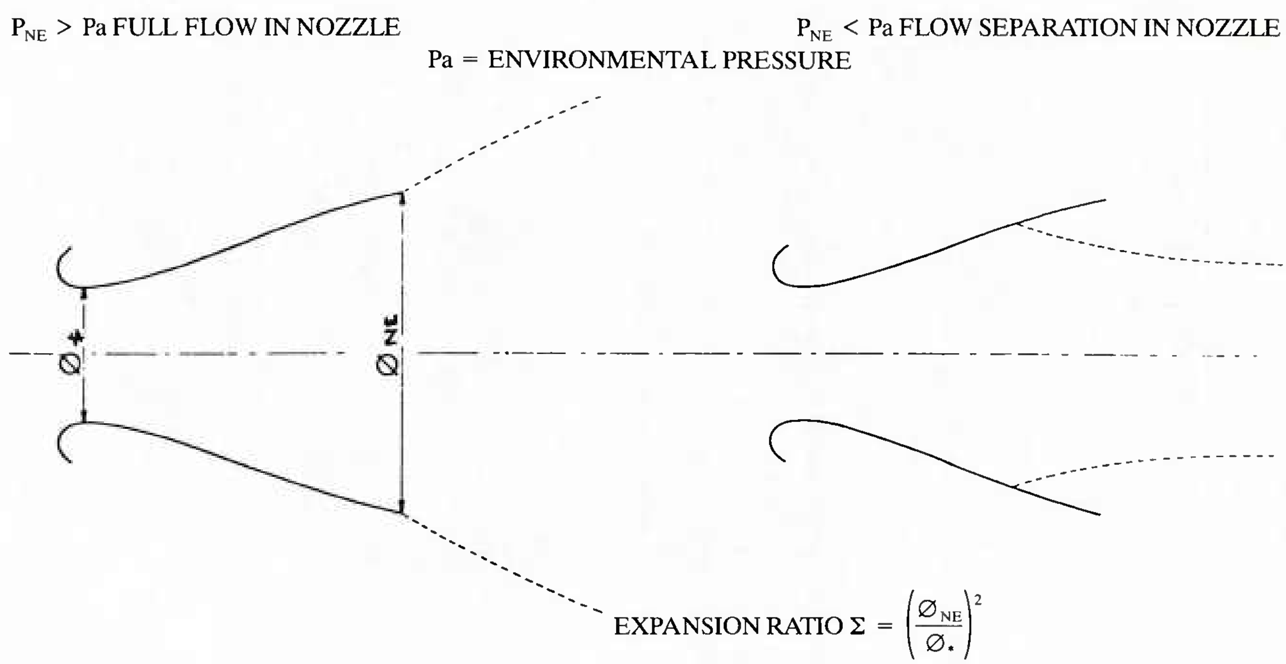
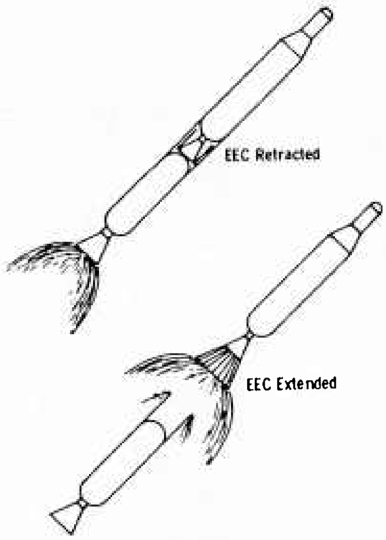


Fig.4

Advantages
- Improved Propulsion System Performance
- No Increase in Vehicle Length or Diameter



Extendable nozzle exit cone

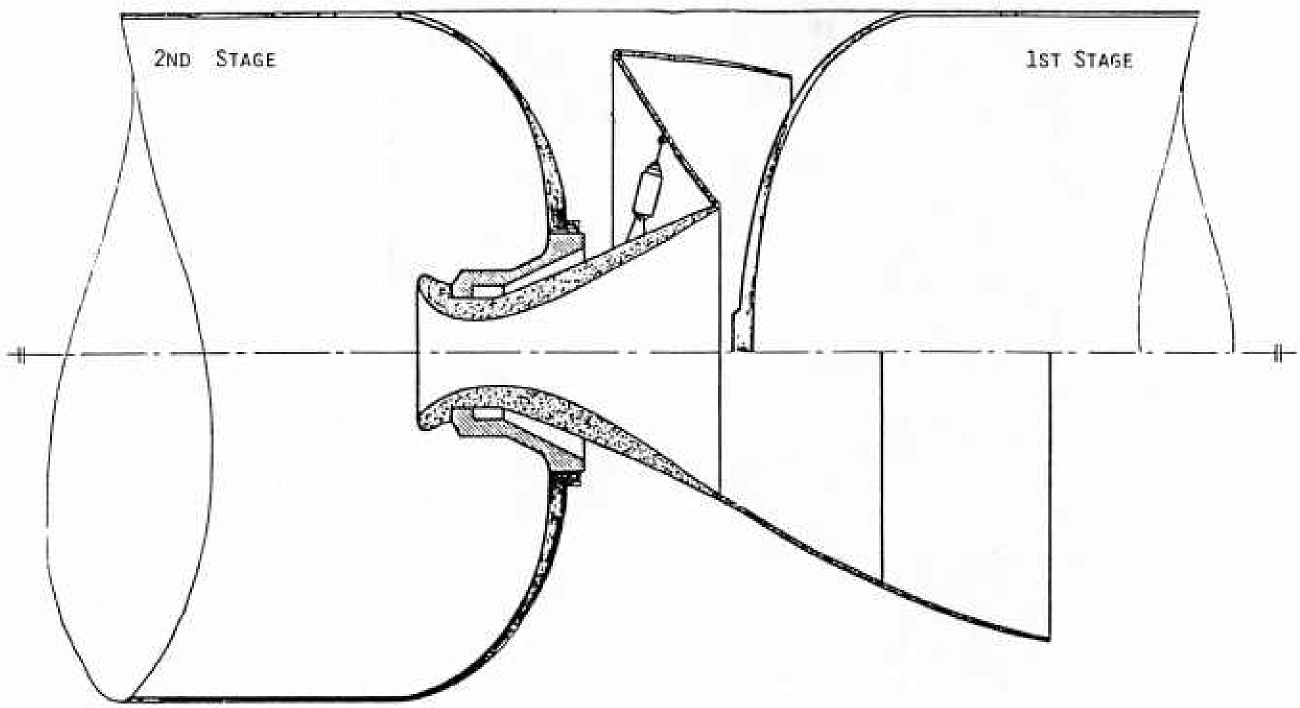
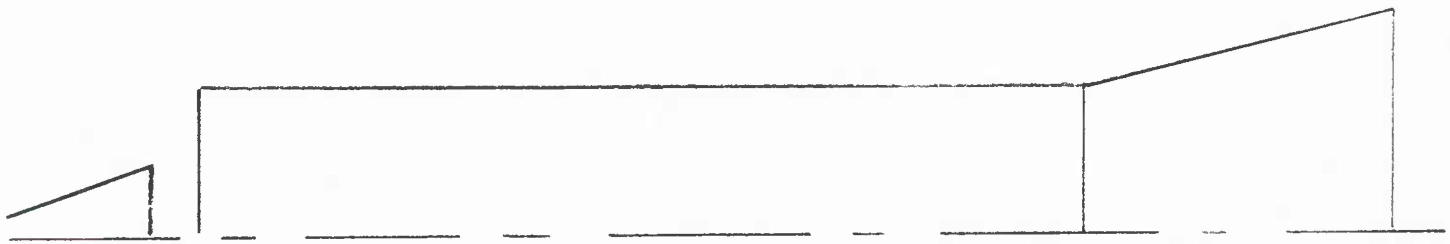
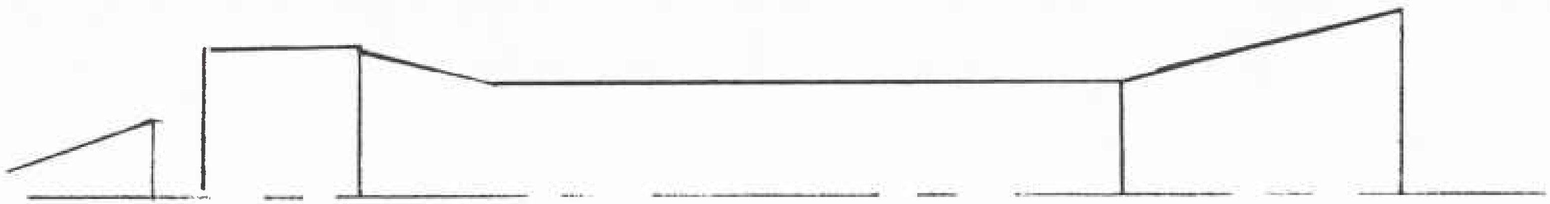


Fig.5 EXTENDABLE EXIT CONE



CYLINDRICAL



SECOND THROAT

Fig.6 SUPERSONIC DIFFUSERS

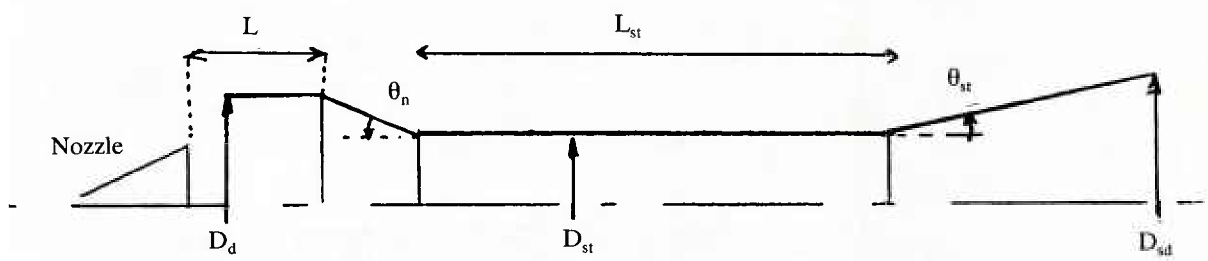


Fig. 7 MAIN DIFFUSER PARAMETERS

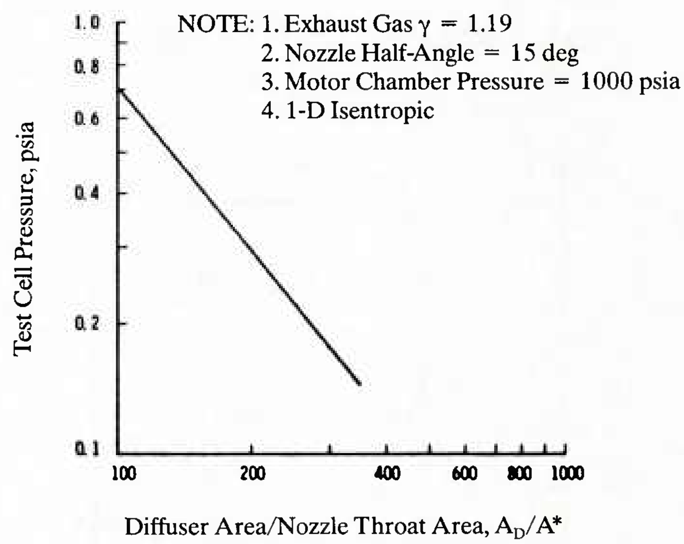


Fig.8 MOTOR PUMPING IN TEST CELL J-5 (AEDC)

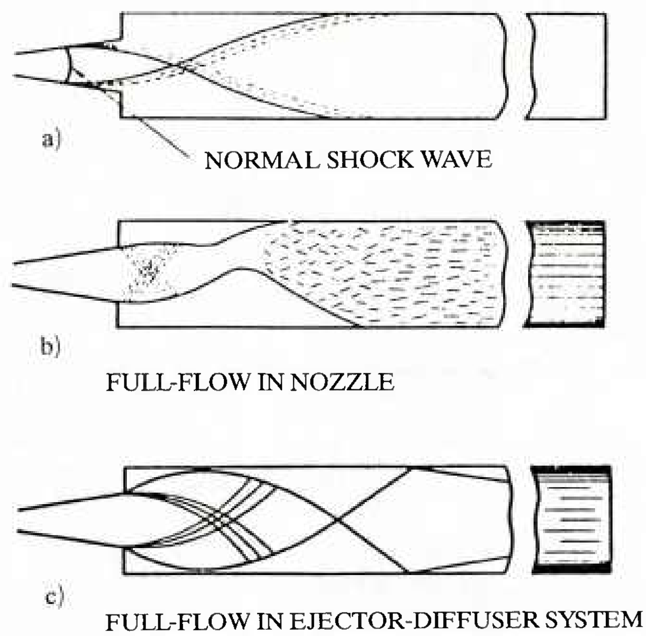


Fig.9 STARTING OF AN EJECTOR-DIFFUSER SYSTEM

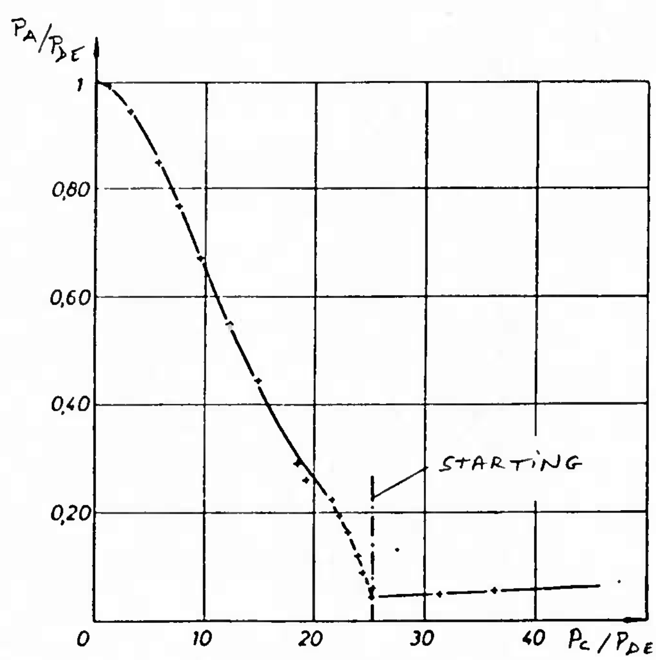
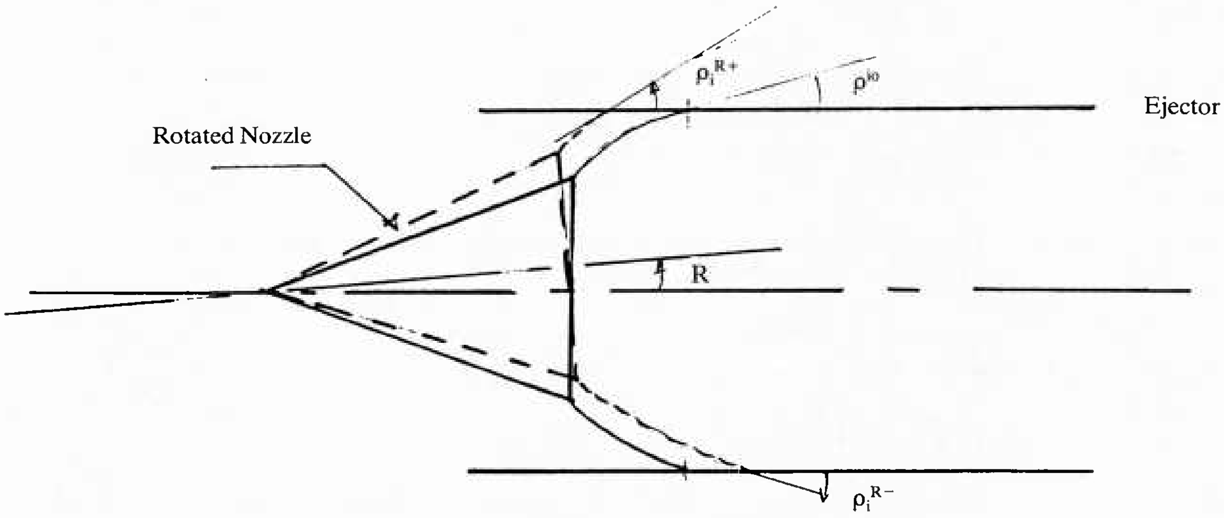
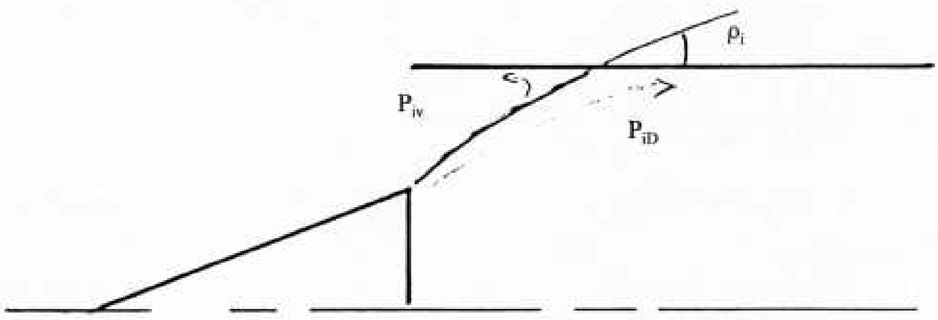


Fig.10 STARTING AND OPERATION VARIATION OF P_A/P_{DE} vs P_C/P_{DE}



R = Rotation
ρ_i = Impingement Angle.
ρ_i^o = Symmetric nozzle Impingement Angle
ρ_i^R = Rotated nozzle Impingement Angles

$$\rho_i^{R-} < \rho_i^o <_i^{R+}$$



P_{iv} = pressure upstream Impingement
P_{iD} = pressure downstream Impingement.

Fig.11

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4.4 UNITS, CONVERSION FACTORS

NAME	S.I. UNITS (A)		ANGLO SAXONS UNITS (B)		CONVERSION FACTOR (C) [*]
Length	Metre	m	Inch Foot	in ft	$2.54 * 10^{-2}$ 0.305
Area	Square Metre	m^2	Square Inch Square Foot	in^2 ft^2	$6.45 * 10^{-4}$ $9.29 * 10^{-2}$
Volume	Cubic Metre	m^3	Cubic Inch Cubic Foot US Gallon	in^3 ft^3 gal	$1.64 * 10^{-5}$ $2.83 * 10^{-2}$ $3.78 * 10^{-3}$
Mass	kilogramme	kg	Pound	lbm	0.454
Force	Newton	N	Pound	lbf	4.448
Temperature	Kelvin Celsius	°K °C	Fahrenheit	°F	°K = (°F + 459.7)/1.8 °C = (°F − 32)/1.8
Energy, Work, Heat Quantity	Joule	j (N*m)	British Thermal Unit	btu	1055.
Power	Watt (J/s)	W	btu/Hr		0.293
Pressure or Stress	Pascal (N/m^2)	Pa	Pound per Square Inch Torr	Psi	6894.8 133.3
Mass Flow Rate		kg/s		lbm/Hr	$1.26 * 10^{-4}$
Density		kg/m^3		lbm/in^3 lbm/ft^3	27680. 16.02
Heat Flux Density		W/m^2		$btu/Hr/ft^2$	3.155

[*] Multiply (B) by (C) to obtain (A)

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